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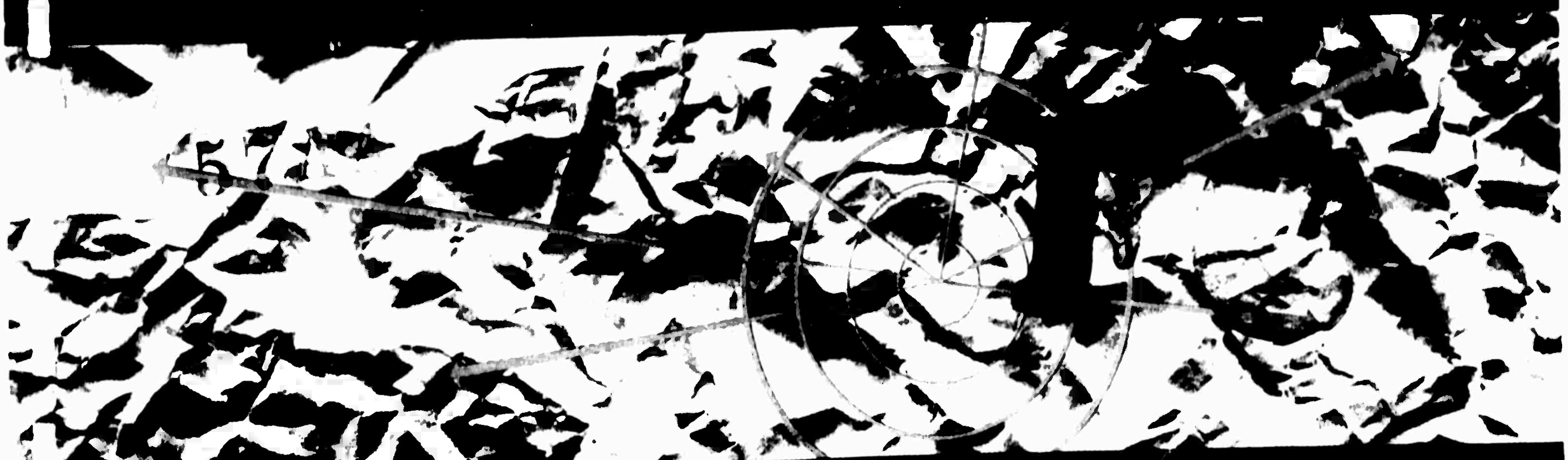
FLYING CRANE TRANSPORTATION SYSTEMS FOR U.S. ARMY



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FLYING CRANE TRANSPORTATION SYSTEMS
1962 - 1967

DUCTED PROPELLER TECHNICAL STUDY

Report ARD No. 124

Contract DA 44-177-TC-382

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ARD 124 CONTRACT DA 44-177-TC-382

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I. INTRODUCTION

The major objective of the present report is to evaluate the potentials of a ducted fan type flying crane. An optimization study is conducted for a series of given missions characterized by payload, range, and hover time. Such a study requires a knowledge of the weights of the various components and of the power requirements in different flight conditions.

Unfortunately, very little information is presently available on the aerodynamic characteristics of a ducted propeller in transverse flow. Truck tests conducted on Hiller's flying platform, see Ref. 1 and Section III,1 of this report, indicate that in forward flight relatively large pitching moments occur which must be compensated by proper means of control. Further, as the moments of inertia of a flying crane about its three principal axes are extremely large, very powerful control moments throughout the speed range are required. The simplest method of generating the necessary control moments in pitch and/or roll is differential collective thrust in a multiple ducted fan configuration. Such a configuration requires a minimum of three ducts. On the other hand, a duct number larger than four is believed to be impractical if a reasonable forward speed must be obtained. This study has, therefore, been limited to a three and four-duct configuration.

As far as possible, performance and control calculations have been based on data derived from experiments. This refers primarily to the hovering power required and to the pitching moments in forward flight. As reliable test data on power required in forward flight are presently not available, theoretical expressions based on the momentum theory have been derived. These theoretical data, in connection with an assumed realistic value for the propeller efficiency, have been used for the power required calculations for all forward flight conditions. To simplify these numerical calculations, general nondimensional charts have been prepared. It should be noted that the additional power required for the compensation of the pitching moments has been taken into account and that interference effects have been neglected. The reason, again, is lack of basic information.

As it is rather difficult to predict, at the present time, the flight characteristics at higher speeds, the cruising speed assumed for the given missions has been arbitrarily limited to 70 knots. This figure is believed to be conservative.

II

POWER AND FUEL REQUIRED

1. Power Required
2. Effect of Flight Duration on Fuel Consumption
3. Fuel/Weight Ratio

G. Sissingh and R. Greenman

II. POWER AND FUEL REQUIRED

1. POWER REQUIRED

Hovering

Theory states that the presence of a duct greatly increases the efficiency of a propeller; experiments conducted so far confirm the theory. The efficiency of a propeller-shroud combination in hovering can best be expressed by the figure of merit, M, defined by the expression

$$\frac{T}{P} = M\sqrt{\frac{2\rho}{T/A}} \tag{1}$$

In this equation

T = thrust, lb

P = power, lb ft/sec

A = propeller disk area, ft^2

T/A = disk loading lb/ft^2

ρ = density of air, lb sec^2/ft^4

For an unshrouded propeller the figure of merit amounts to approximately M = 0.7 to 0.75; for a properly designed propeller shroud combination this value goes up to approximately 1.5. According to equation (1) this means that for given power and propeller diameter the ducted propeller produces up to 60% more static thrust than a conventional unshrouded propeller.

In order to derive a realistic value for the anticipated figure of merit of a ducted-fan type Flying Crane, a survey of the test data available has been conducted. Fortunately, already a considerable amount of static testing has been done. Some of the results are discussed in the following paragraphs.

Fig. 1, derived from test data reported in Ref. 3, shows the figure of merit M of a shrouded and unshrouded propeller against the blade pitch setting. The maximum figure of merit of the shrouded configuration amounts to approximately 1.15. It should be noted, however, that this propeller-shroud combination has been laid out for an advance ratio of

0.95. It may, therefore, be expected that by using a duct form which favors the low speed range, higher figures of merit for the hovering condition can be obtained. This is confirmed by the curve shown in Fig. 3. This curve, plotted against the coefficient C_T as defined in the original NACA report, represents the figure of merit of the "short-cruise" shroud tested by R. J. Platt, see Ref. 2. According to Fig. 3, in this case values of M = 1.5 and higher are obtained.

In Fig. 2 the figure of merit of various other test data is plotted against the disk loading. These data come from different sources. The upper curve represents tests conducted by the Doak Aircraft Company, Ref. 8. The two lower curves are taken from Ref. 5, they are the results of a survey made by A. Stone, BuAer, and refer to an area ratio of 1.0 and 1.2, respectively. Finally, the single point plotted in Fig. 2, is taken from Ref. 7. The various data represented in Fig. 2 fall into the range $1.23 < M < 1.56$ where the lower limit is partly based on Krueger's tests, which, as mentioned previously, have been conducted on propeller-shroud combinations laid out for high advance ratios. It appears, therefore, that by a proper design, at least values of M = 1.3 to 1.4 can be obtained. For the hovering performance calculations of the Flying Crane M = 1.31 has been assumed, this figure is believed to be conservative. In the preliminary design studies of this report, the engines are located in the center of the ducts, and transmission losses are, therefore, relatively low. It has been assumed that these losses amount to approximately 2.5%, i.e., the transmission efficiency $\eta_t = 0.975$. With these assumptions it follows from equation (1) that the total hovering power required amounts to

$$(HP)_{hovering} = \frac{W}{550M\eta_t}\sqrt{\frac{w_c}{2\rho}} \qquad (2)$$

where

W = gross weight, lb

w_e = effective disk loading, lb/ft^2

and

$$M\eta_t = 1.31 \times 0.95$$
$$= 1.28$$

Forward Flight

As mentioned previously, no test data are presently available on forward flight characteristics, i.e., on power required in transverse flow conditions. The performance calculations of level forward flight and climb have therefore been based on equations derived from the momentum theory. Compressibility effects have been neglected.

For simplicity, at present only one ducted propeller is considered. The equations can also be applied directly to a multiple ducted fan configuration if power required, weight, and external drag are interpreted as power required per ducted propeller, weight carried per ducted propeller, and drag per ducted propeller.

If no additional means of propulsion and lift generation are used, in level flight the vertical component of the net thrust vector must be equal to the weight and the horizontal component equal to the external drag. Let be

W = weight, lb

D_e = external drag, lb

D_i = internal drag (acting in the direction of duct axis), lb

V_o = flight velocity, ft/sec

V_e = duct exit velocity, ft/sec

A_e = duct exit area, ft^2

m = mass flow per second, lb sec/ft

If α denotes the forward tilt angle of the duct axis and

$$T = W \hat{+} D_e \hat{+} D_i \tag{3}$$

the resultant force vector, it follows from Fig. 4 that the horizontal component of T must be equal to $(D_e + D_i \sin\alpha)$, and the vertical component equal to $(W + D_i \cos\alpha)$.

This means that the following equation must be fulfilled

$$T^2 = (D_e+D_i \sin\alpha)^2 + (W+D_i \cos\alpha)^2 \tag{4}$$

On the other hand, the momentum theory states

$$T = mV_o \hat{-} mV_e \tag{5}$$

or, from Fig. 4,

$$T^2 = m^2V_o^2 + m^2V_e^2 - 2m^2V_oV_e \sin\alpha \tag{6}$$

Equating the right hand sides of equation (4) and equation (6) leads to

$$m^2V_o^2 + m^2V_e^2 - 2m^2V_oV_e\sin\alpha = W^2 + D_e^2 + D_i^2 + 2WD_i\cos\alpha + 2D_eD_i\sin\alpha \tag{7}$$

where the mass flow

$$m = V_eA_e\rho \tag{8}$$

From Fig. 4 the following equations for the required duct tilt angle can be derived:

$$\sin\alpha = \frac{D_e + mV_o}{mV_e - D_i} \tag{9}$$

$$\cos\alpha = \frac{W}{mV_e - D_i} \tag{10}$$

$$\tan\alpha = \frac{D_e + mV_o}{W} \tag{11}$$

The theoretical studies can greatly be simplified by introducing non-dimensional coefficients. Let be

$$\Psi = \frac{W/A_e}{\rho V_o^2} \tag{12}$$

$$\varepsilon = V_e/V_o \tag{13}$$

$$f_e = \frac{D_e}{A_e V_o^2 \rho/2} \tag{14}$$

$$f_i = \frac{D_i}{A_e V_e^2 \rho/2} \tag{15}$$

The most significant of these nondimensional coefficients is the quantity Υ, which determines the aerodynamic characteristics of a given flight condition. It can easily be seen from equation (12) that 2Υ can be interpreted as a conventional lift coefficient referred to the wing area A_e and the free-stream velocity V_o. The parameter Υ should be considered as the major parameter of a ducted fan, for this reason the various quantities which determine power required, tilt angle, pitching moment, etc., have later been calculated and plotted as function of Υ. It may be of interest to note that for a Flying Crane, as investigated in this report, the quantity Υ falls into the range $1 < \Upsilon < \infty$ where $\Upsilon = \infty$ refers to the hovering condition. See also Fig. 5 where Υ is plotted vs disk loading for several velocities. These curves refer to S.L. conditions.

Another important parameter is the quantity ε which, according to equation (13), represents the ratio (duct exit velocity)/(free stream velocity). Finally, f_i and f_e characterize the internal and external drag of a ducted propeller configuration and can be interpreted as drag coefficients. It should be noted that the external drag coefficient f_e is referred to the free stream velocity V_o, and the internal drag coefficient f_i to the duct exit velocity V_e. For the disk loadings and the speed range of a ducted-fan type Flying Crane or, more appropriately, for its Υ-range, the internal drag is of minor importance. Evaluation of test data and preliminary numerical studies show that values of approximately $f_i = 0.08$ to 0.095 must be expected. The performance calculations of this report have conservatively been based on

$$f_i = 0.1 \tag{16}$$

With equations (12), (13), (14), (15) the equations (9), (10), (11) simplify to

$$\sin\alpha = \frac{f_e + 2\varepsilon}{2\varepsilon^2 - f_i} \tag{17}$$

$$\cos\alpha = \frac{2\Psi}{2\varepsilon^2 - f_i} \tag{18}$$

$$\tan\alpha = \frac{\varepsilon + 1/2\, f_e}{\Psi} \tag{19}$$

Similarly, equation (7) can be reduced to

$$\varepsilon^4 \left(1 - 1/2\, f_i\right)^2 - \varepsilon^2 - \varepsilon f_e = \Psi^2 + 1/4\, f_e^{\,2} \tag{20}$$

The last equation permits the calculation of ε as function of disk loading, speed, external and internal drag. As the knowledge of this quantity is mandatory for several reasons (determination of tilt angle, power required) general charts have been prepared which will be discussed later.

The momentum theory states that for the ideal case (propeller and transmission efficiency = 1) the power required amounts to

$$(\text{power})_{\text{ideal}} = \frac{m}{2} \left(V_e^{\,2} - V_o^{\,2}\right) \tag{21}$$

where the mass flow is given by equation (8). If η_p, η_t denote the propeller and transmission efficiency, respectively, the brake HP required for level flight becomes

$$(\text{HP})_{LF} = \frac{m}{2 \times 550\, \eta_p \eta_t} \left(V_e^{\,2} - V_o^{\,2}\right) \tag{22}$$

With the nondimensional coefficients given by equations (12), (13) the above equation can be rewritten as

$$(\text{HP})_{LF} = \frac{W V_o}{2 \times 550\, \eta_p \eta_t} \times \frac{\varepsilon(\varepsilon^2 - 1)}{\Psi} \tag{23}$$

Similar to equation ([illegible]), which determines the power requirement for hovering, equation (23) can also be expressed as

$$(HP)_{LF} = \frac{W}{550\tau\eta_p\eta_t}\sqrt{\frac{w_e}{2\rho}} \tag{24}$$

where the nondimensional quantity τ represents a kind of figure of merit for forward flight. Comparison of equations (23), (24) gives

$$\tau = \frac{\sqrt{2}\Psi^{3/2}}{\varepsilon(\varepsilon^2-1)} \tag{25}$$

The numerical performance calculations have been based on the following assumptions, believed to be realistic

$$\begin{aligned} \eta_p &= 0.87 \\ \eta_t &= 0.975 \\ \therefore \eta_p\eta_t &= 0.85 \end{aligned} \tag{26}$$

In order to simplify the numerical investigations several charts have been prepared, see Figs. 6 to 11. The curves represent the quantities ε, α, $\varepsilon(\varepsilon^2-1)$, and τ, plotted against the parameter Ψ. As can be seen from Fig. 5, for the assumed cruising speed of 70 knots and for the disk loadings investigated, the parameter Ψ lies within the limits $1 < \Psi < 10$. Therefore, the curves represented in Figs. 6 to 11 are in most cases restricted to this Ψ-range. Inspection of the functions represented in these graphs leads to the following conclusions.

Figs. 6 and 7 show the velocity ratio ε for $f_i = 0$ and $f_i = 0.1$, respectively, for an external drag corresponding to $f_e = 0$, .24 and .48. Comparison of these curves indicates that within the range investigated the internal drag has only a minor effect and that, as expected, the effect of the external drag increases with decreasing Ψ-values, i.e., with increasing speed.

Fig. 8 shows the duct-tilt angle α vs Ψ for various external drag coefficients. The internal drag is assumed to be $f_i = 0.1$, slip-stream deflection by vanes is not taken into account. The curves of Fig. 8 show that even for zero external drag appreciable forward tilt angles are required. For instance, at a disk loading of 50 lb/ft^2 and a speed of 70 knots, the parameter Ψ is approximately 1.5. According to Fig. 8 a tilt angle of approximately $\alpha = 45^\circ$ is required for this flight condition. This figure refers to zero external drag, it increases slightly if external drag is considered.

Figs. 9, 10 show the function $F = \varepsilon(\varepsilon^2-1)$ and its first derivative $F' = dF/d\Psi$. The former function plays a roll in the calculation of power required for forward flight. The latter is needed later to calculate the variation of power required caused by changes in weights due to fuel consumption. For $\Psi < 1$ the following approximation can be used

$$F' = \frac{1+6x}{4x}\sqrt{x-\frac{1}{2}} \qquad (27)$$

where

$$x = \sqrt{\Psi^2+\frac{1}{4}} \qquad (28)$$

Fig. 10 shows F' as given by equation (27), it represents the mathematically correct solution for the simplified case $f_i = f_e = 0$.

In Fig. 11 the parameter τ is represented which, according to equation (24), determines the power required for level flight. The curve is based on the following assumptions

$$f_i = 0.10$$

$$f_e = 0.36$$

The justification for the selection of the above f_e-value will be discussed later. It may be worthwhile mentioning, however, that within the speed range investigated a 20% in - or decrease in the external drag has only a minor effect on the power required.

Evaluation of external drag parameter f_e

It is estimated that the equivalent parasite area of the aircraft, without load, corresponds to that of a rectangle with the length 2.5D, and the width 0.3D. The equivalent parasite area of the load is given as 80 ft². This means that, by definition,

$$f_e = \frac{0.75D^2 + 80}{b\pi D^2/4} \qquad (29)$$

where

D = propeller diameter
b = number of ducted propellers

The drag parameter f_e as given by equation (29) is plotted in Figure 12 against the propeller diameter D. Both the 3 and 4 duct configurations are shown. In each case f_e decreases with increasing D-values. Also plotted is the value $f_e = 0.36$ on which the performance calculations of this report have been based. For a 4-duct configuration with $D > 15$ ft the drag parameter is considerably lower than $f_e = 0.36$. On the other hand, for the 3-duct configuration with small propeller diameters f_e is somewhat higher than 0.36. As mentioned previously, for the speed range considered in this report ($V < 70$ knots), the external drag has only a minor effect on the power required. It is, therefore, believed that the assumption of a constant f_e-value is justified and within the limits of the accuracy with which the performance can be predicted today.

Climb

For climbing flight (γ = angle of climb) the forces acting parallel and normal to the flight path area

in direction of flight: Drag + (Weight x sin γ)

normal to direction of flight: W cos γ

This means that the nondimensional coefficients Ψ, f_e (referring to level flight) for climbing flight change to

$$\Psi_c = \Psi \cos \gamma \tag{30}$$

$$(f_e)_c = f_e + 2\Psi \sin \gamma \tag{31}$$

The total power required for climbing can again be calculated by equation (23) if Ψ and f_e are replaced by Ψ_c, and $(f_e)_c$, respectively. The power required can also be expressed as

$$(HP) = \frac{W\,V_o}{2 \times 550 \eta_p \eta_t} (\zeta + \Delta\zeta) \tag{32}$$

where the first term in the parantheses refers to the power required in level flight and the second to the excess power required for climbing, i.e.

$$\frac{\Delta\zeta}{\zeta} = \frac{\text{Excess power required for climbing}}{\text{Power required for level flight}} \tag{33}$$

The excess power required for climbing can also be written as

$$(HP)_c = \frac{W\,V_c}{550 \eta_c \eta_p \eta_t} \tag{34}$$

where the rate of climb

$$V_c = V_o \sin \gamma \tag{35}$$

The term η_c in equation (34) represents the climbing efficiency; from equations (32) and (34) it follows that

$$\eta_c = \frac{2 \sin \gamma}{\Delta \xi} \tag{36}$$

In Figure 13 the climbing efficiency η_c as defined by equation (34) has been plotted against the power ratio $\Delta\xi/\xi$. Curves for Ψ = 4, 8, 12, 16, and 20 are shown, also plotted are curves for constant γ values. The various curves in Figure 13 indicate that the climbing efficiency decreases with increasing climb angles γ and increasing Ψ-values.

Figure 13 can be used to calculate the rate of climb as follows. Let be

$(HP)_{AV}$ = Horsepower available

$(HP)_{LF}$ = Horsepower required for level flight

$$\frac{\Delta\xi}{\xi} = \frac{(HP)_{AV} - (HP)_{LF}}{(HP)_{LF}}$$

From equation (34) it follows that the rate of climb

$$V_c = \left\{ (HP)_{AV} - (HP)_{LF} \right\} \times \frac{550\, \eta_c \eta_t \eta_p}{W} \tag{37}$$

where η_c can be taken from Figure 13 as function of $\Delta\xi/\xi$ and Ψ. It may be worthwhile mentioning that for the flying crane study of this report, due to the hovering requirement (6000 ft, 95°), the ratio $\Delta\xi/\xi$ for an altitude of 2000 ft amounts to approximately 0.4.

2. EFFECT OF FLIGHT DURATION ON FUEL CONSUMPTION

Hovering

Let be

$(HP)_o$ = power required at the beginning of hover period

SFC = specific fuel consumption lb/HP/hour

t_H = hover time, minutes

Differentiation of the basic equation (2) for power required in hovering gives

$$\frac{d(HP)}{dW} = \frac{3}{2x550M\eta_t}\sqrt{\frac{w_e}{2\rho}} \qquad (38)$$

A first approximation for the change in weight due to fuel consumption is

$$dW = \frac{(HP)_o x(SFC)x t_H}{60} \qquad (39)$$

Inserting equation (39) into equation (38) leads to

$$d(HP) = \frac{(HP)_o(SFC)t_H}{40x550M\eta_t}\sqrt{\frac{w_e}{2\rho}} \qquad (40)$$

This means that at the end of the hover time the power required amounts to

$$(HP)_{t_H} = (HP)_o\left\{1 - \frac{(SFC)t_H}{40x550M\eta_t}\sqrt{\frac{w_e}{2\rho}}\right\} \qquad (41)$$

and that the average power required during the hover period is approximately

$$(HP)_{average} = HP_o\left\{1 - \frac{(SFC)t_H}{80x550M\eta_t}\sqrt{\frac{w_e}{2\rho}}\right\} \qquad (42)$$

Based on this average power required, the fuel consumption in lbs for a given hover time t_H in minutes becomes

$$\text{Fuel Weight} = \frac{(SFC)t_H W_c}{60x550M\eta_t}\sqrt{\frac{w_e}{2\rho}}\left\{1 - \frac{(SFC)t_H}{80x550M\eta_t}\sqrt{\frac{w_e}{2\rho}}\right\} \qquad (43)$$

Forward Flight

An analogous expression can be derived for forward flight. If $(HP)_o$ denotes again the power required at the beginning of the cruise, a first approximation for the decrease in weight due to fuel consumption is given by

$$dW = (HP)_o(SFC)\text{ time} \qquad (44)$$

The time required to travel the range R (nautical miles) at the speed V_o (ft/sec) is

$$\text{Cruising time} = 1.69\ R/V_o \text{ hours} \tag{45}$$

Inserting equation (45) into equation (44) gives

$$dW = \frac{(HP)_o (SFC)\ 1.69R}{V_o} \tag{46}$$

From

$$\frac{d\Psi}{\Psi} = \frac{dW}{W} \qquad \text{(See equation (12))} \tag{47}$$

$$\frac{d(HP)}{(HP)_o} = \frac{dF}{F} \qquad \text{(See equations (12), (23))} \tag{48}$$

$$dF = F'd\Psi \qquad \text{(by definition)} \tag{49}$$

it follows

$$\begin{aligned} d(HP) &= (HP)_o \frac{F'}{F} \frac{dW}{W} \Psi \\ &= (HP)_o \frac{F'}{F} \frac{dW}{W} \frac{w_e}{\rho V_o^{\ 2}} \end{aligned} \tag{50}$$

With dW as given by equation (46), the above equation (50) can be rewritten

$$d(HP) = (HP)_o^{\ 2} \frac{F' w_e (SFC) 1.69R}{F \rho V_o^{\ 3} W} \tag{51}$$

This means that the power required at the end of the cruise amounts to approximately

$$(HP)_o - d(HP) = (HP)_o \left\{ 1 - \frac{F'(SFC) 1.69R}{2x550} \right\} \tag{52}$$

where the function F' can be taken from Figure 10. The average power required during the cruise period is

$$(HP)_{average} = (HP)_o \left\{ 1 - \frac{F'(SFC) 1.69R}{4x550} \right\} \tag{53}$$

which means that during this cruising period the following amount of fuel is consumed

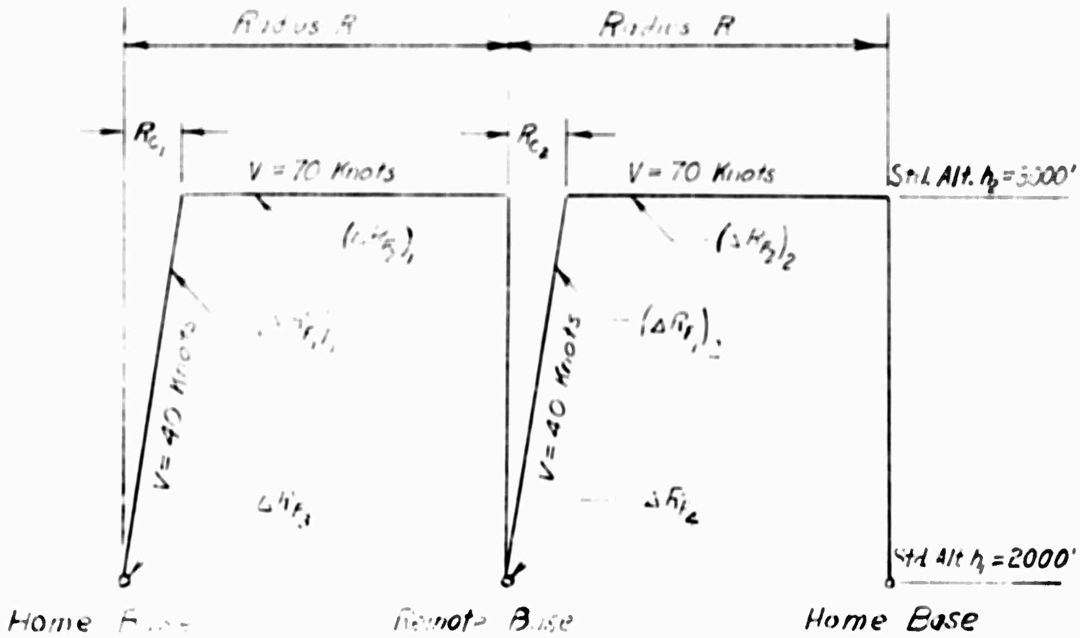
$$\text{Fuel weight} = (HP)_{average} x (SFC) x (time)$$

$$= \frac{(SFC)R}{V_{cr}} (HP)_o \left(1 - \frac{F \cdot (SFC) 1.69R}{4x550}\right) \tag{54}$$

In this equation, which gives the fuel consumption in lbs, V_{cr} denotes the cruising speed in knots and R the range in nautical miles. The term in the parantheses represents the average reduction in fuel consumption or power required due to the decreasing weight.

3. FUEL-WEIGHT RATIO

The various basic equations for power required, fuel consumption, etc., derived in the previous sections are applied to a specific mission which is described schematically.



This mission consists of the following operations under standard atmospheric conditions:

1. Warming up at 100% normal rated power at home base with full load at 2000 foot altitude.

2. Climbing from 2000 to 3000 feet.

3. Cruising at 3000 feet to remote base at distance R from home base.

4. Hovering at 2000 feet with release of payload.

5. Climbing to 3000 feet.

6. Cruising back to home base at 3000 feet.

7. Carrying a fuel reserve of ten percent of initial fuel.

The specific fuel consumption (SFC), that is employed in all computations, is assigned a value of .55 lbs/BHP-hr at 100% normal rated power (NRP) and at sea level standard day and made proportional to the SFC characteristics of the gas turbine engine of Chart II of Hiller Report No. 630.5, see Reference 9. The SFC versus NRP curves are presented in Figure 14. The installed power is given by the hovering capability at 6000 feet altitude and 95°F day. NRP available at different altitudes is made proportional to the NRP of the above mentioned Hiller report. The NRP versus altitude curve is also presented in Figure 14.

The general equation for the dimensionless ratio R_F of fuel required to gross weight may be written as follows:

$$R_F = 1.10 \left[(\Delta R_{F_1})_1 + (\Delta R_{F_2})_1 + \Delta R_{F_3} + \Delta R_{F_4} + (\Delta R_{F_1})_2 + (\Delta R_{F_2})_2 \right] \quad (55)$$

In the above equation, which is patterned after the presentation in Reference 10, the various increments in R_F refer to parts of the general mission.

$(\Delta R_{F_1})_1$ is the fuel to weight ratio for climb from altitude h_1 = 2000 feet to h_2 = 3000 feet on a standard day at a speed of 40 knots and normal rated power.

It can easily be seen that

$$(\Delta R_{F_1})_1 = \frac{(h_2 - h_1)}{(60)(R/C)} (SFC) \left(\frac{BHP}{W_G}\right) = \frac{(1000)}{(60)} \frac{(SFC)}{(R/C)} \left(\frac{BHP}{W_G}\right) \quad (56)$$

where W_G denotes the design gross weight, and BHP the total power required in climbing flight. BHP is the sum of the expressions given by equations (24), (34). For the calculation of the effective disk loading, w_e, in equation (24) it has been assumed that only 91% of the area is effective which means

$$w_e = \frac{1.1W}{b\pi D^2/4} \quad (57)$$

The rate of climb

$$R/C = 33000\eta_c \left[\frac{AHP}{W} - \frac{BHP_{LF}}{W}\right] \text{ ft/min} \quad (58)$$

where

AHP = available horsepower at 2500 feet, std. day

$= 1.460 \ AHP_{6000', \ 95^\circ \ day}$

As mentioned previously, for the Flying Crane studies of the present report the ratio (excess power available for climb)/(level flight power) is approximately 40%. For this particular case a conservative approximation for the rate of climb is given by

$$R/C = 7.6\ V_o \sqrt{\Psi}\ \text{ft/min} \tag{59}$$

where V_o is the flight velocity in ft/sec (assumed to be 40 knots = 68 ft/sec) and Ψ is defined by equation (12).

$(\Delta R_{F_1})_2$ is also given by equation (56) where for the calculation of the power required, BHP, the reduced weight due to fuel consumption and due to the released load has to be taken into account.

$(\Delta R_{F_2})_1$ and $(\Delta R_{F_2})_2$ represent the fuel to weight ratios for cruising. The cruising speed is assumed to be 70 knots. According to equations (53), (67) the average HP required for cruising amounts to

$$BHP = \frac{W_o k_c}{550\tau\eta} \sqrt{\frac{w_e}{2\rho}} \left[1 - \frac{F'(SFC)(R-R_c)}{1300}\right] \tag{60}$$

This means

$$(\Delta R_{F_2})_{1,2} = \frac{(SFC)(BHP)(R-R_c)}{V_{cr}\ W_G} \tag{61}$$

In these equations

W_o = actual weight at the beginning of the cruise

W_G = design gross weight

R = design radius of action, naut. mi.

R_c = range credit during climb, naut. mi.

V_{cr} = cruise speed, in knots

ΔR_{F_3} is the fuel to weight ratio for a starting time of 2 minutes under condition of expenditure of 100% normal rated power.

$$\Delta R_{F_3} = \left(\frac{t_s}{60}\right)(SFC)\left(\frac{BHP}{W_G}\right) = \left(\frac{2}{60}\right)(SFC)\left(\frac{BHP}{W_G}\right) \tag{62}$$

ΔR_{F_4} is the fuel to weight ratio for hovering at altitude h_1 = 2000 feet with full load.

$$\Delta R_{F_4} = (SFC)\left(\frac{t_H}{60}\right)\left(\frac{BHP}{W_G}\right) \tag{63}$$

In the above equation, BHP represents the average power required for hovering

$$BHP = \frac{W_o}{550M\eta_t}\sqrt{\frac{W_e}{2\rho}}\left[1 - \frac{(SFC)(t_H)}{44,000}\sqrt{\frac{W_e}{2\rho}}\right] \tag{64}$$

where

t_H = mission hover time, minutes

ρ = air density, .002242 slugs/cu.ft

III. CONTROL CONSIDERATIONS

1. METHODS OF CONTROL

It is assumed that control about the longitudinal and lateral axis is achieved by differential propeller thrust. As a variation in thrust also affects the torque, the use of counterrotating propellers is mandatory for the three-duct configuration. If the directions of rotation are properly chosen, for the four-duct configuration also single propellers can be employed. However, in order to avoid large gyroscopic moments due to angular velocities in pitch or roll, the design studies of both configurations have been based on counterrotating propellers. See also Drawing Nos. 1 and 2, which show a 3-view sketch of each configuration.

As there is no need for large angular accelerations in yaw, it is believed that yaw control can best be achieved by differential slip-stream deflection, preferably by vanes arranged in the fore-aft direction. In order to produce a pure yawing moment, the force to the left must be equal to that to the right. This means that for the 3-duct configuration, the single duct in the front requires about the same vane area of those of the other two combined. It will be seen later that in forward flight relatively large nose-up pitching moments occur. In order to compensate these moments, the thrust of the rear propeller(s) must be increased and that of the front propeller(s) decreased. This fact, together with considerations relating to the static stability in forward flight, determined the duct arrangement of the 3-duct configuration which has one duct in the front and two in the rear.

2. EFFECT OF CONTROL ON POWER REQUIRED

As mentioned previously, in forward flight relatively large nose-up pitching moments occur which must be compensated by differential thrust. The pitching moment per duct can be expressed as:

$$M = C_m AqD \tag{65}$$

where $C_m = f(\Psi)$ is a nondimensional pitching moment coefficient and

A = propeller disk area, ft^2

D = propeller diameter, ft

q = dynamic pressure, lb/ft^2

If b denotes the number of ducts, the total pitching moment amounts to approximately

$$M_{total} = bC_m AqD \tag{66}$$

This pitching moment depends, of course, on the c.g. location of the Flying Crane. A first approximation can be obtained from the C_m-curves plotted in Figure 15.

The increase in total power required is taken into account by adding a factor $k_c > 1$ to the performance equation (24) which thus becomes

$$(HP)_{LF} = \frac{Wk_c}{550\eta\pi} \sqrt{\frac{w_e}{2\rho}} \tag{67}$$

In this equation

$$\eta = \eta_p \eta_t \tag{68}$$

denotes the overall efficiency assumed to be 0.85. The numerical calculations have been based on $k_c = 1.04$ which is approximately the maximum found at a speed of V = 70 knots. The figure $k_c = 1.04$ states that a 4% increase in total power is necessary to produce the differential thrust required for pitch control. It should be noted that the changes for the individual propellers are considerably higher. For the rear-propellers, which have to produce a larger thrust, the increase amounts up to approximately 25% for the 3-duct configuration and up to 33% for the 4-duct configuration. This can best be shown by the following example which is typical.

<u>Example</u>:	3-Duct Configuration	60 knots, SL
	Gross Weight	96,000 lbs
	Duct Diameter	28.6 ft
	Fore-aft distance between Ducts	36.5 ft

Without consideration of control moments:

	Lift per Propeller	32000 lbs
	Power per Propeller	5670 HP
	Total Power	17010 HP

With consideration of control moments:

The pitching moment amounts to 432,000 lb.ft which means that the lift of the front propeller has to be decreased by 11,800 lbs and that of each rear propeller is increased by 5900 lbs. The resulting change in lift and power distribution is shown in the following table.

	Lift, lbs	Power, HP
Front Propeller:	20.200	3100
Each Rear Propeller:	37.900	7200
Total:	96.000	17500

In this case the increase in total power amounts to approximately 3%; it is believed that by the development of proper duct shapes the effect of control on power required can be minimized. However, until this information is available, the additional losses of approximately 4% at 70 knots should be taken into account.

IV. DERIVATION OF WEIGHT EQUATIONS

1. GENERAL METHOD

In order to determine the empty weight of the aircraft, the weights of the components are first derived. The components are listed as follows:

Rotor Weight	W_R
Transmission Weight	W_T
Duct Weight	W_D
Engine Weight	W_E
Engine Accessory Weight	W_{EA}
Structural Weight (Beams)	W_{SB}
Structural Weight (Pylons)	W_{SP}
Structural Accessory Weight	W_{SA}
Other Weight	W_O

The ratio of aircraft empty weight to gross weight is called ϕ.

$$\phi = \Sigma \text{ Component weights/Gross weight, } W_G$$

The weight of fuel and fuel tanks equals gross weight less payload, W_P, and empty weight, W_{empty}.

$$W_{fuel} + W_{tanks} = W_G - W_P - W_{empty}$$

The weight of fuel tanks is a constant proportion of fuel weight. Therefore, fuel weight can be expressed as a constant, K, times weight of fuel plus tanks.

$$W_{fuel} = K\,(W_{fuel} + W_{tank}) = K\,(W_G - W_P - W_{empty})$$

Dividing this equation by W_G gives:

$$\frac{W_{fuel}}{W_G} = K\left(\frac{W_G}{W_G} - \frac{W_P}{W_G} - \frac{W_{empty}}{W_G}\right)$$

or

$$R_F = K\left(1 - \frac{W_P}{W_G} - \phi\right)$$

The ratio of fuel weight to gross weight, R_F, is called the "Fuel Available Ratio". It is equated to the "Fuel Required Ratio" to determine the gross weight of a design that will satisfy a given set of conditions. In solving this equation R_F is plotted against W_G and disk loading, w. Therefore, W_G and w are considered as independent variables and all component weights are determined in terms of them.

Both analytical and statistical methods are used in deriving the component weight expressions. Values of each component weight have been tabulated in order to obtain tables of values for ϕ, from which in turn are obtained values of fuel available. The component weights have been tabulated over a wide enough range of W_G and w to include the intersections of the Fuel Available curves with the Fuel Required curves.

In order to visualize various configurations and to make some design sketches, it is necessary to know rotor diameters and powerplant sizes. These are calculated below.

Rotor Diameter, D

$$D = \sqrt{\frac{4W_G}{\pi b w}}$$

D = diameter, ft

b = No. ducts

W_G = gross weight, lb

w = disk loading, psf

For

W_G = 25000 lbs

w = 35 psf

b = 3

$$D = \sqrt{\frac{4 \times 25000}{\pi \times 3 \times 35}} = 17.4 \text{ ft}$$

2. COMPONENT WEIGHTS

2.1 Rotor Weight, W_R

Reference (12) gives the following expression for the weight of a Curtiss propeller for conventional, fixed-wing aircraft:

$$\text{Prop. Weight} = K\left(\frac{AF}{100}\right)^{1.8} D^4 N^2 B^{.825}$$

Where

AF = activity factor

D = diameter, ft

N = take-off rpm

B = number of blades

$K = .26 \times 10^{-8}$ for turbo-props

$= .231 \times 10^{-8}$ for recip. engine props.

The accuracy of this formula is checked against data from the Curtiss catalog, Reference (13).

Check No. 1 Curtiss 634S - C500, 1052, 3 blade, steel, single rotation, for reciprocating engine.

$D = 16.67$ ft

$AF = 113$

T.O. rpm = 1225

Actual weight = 699 lb

$$\text{Calculated weight} = .231 \times 10^{-8}\left(\frac{113}{100}\right)^{1.8}(16.67)^4(1225)^2(3.)^{.825}$$

= 753 lb (This is within 8 percent)

Check No. 2 Curtiss CG44S - B400, 830, 4 blade, steel, for reciprocating engine

$D = 15.1$ ft

$AF = 120$

T.O. rpm = 1260

Actual weight = 859 lbs

$$\text{Calculated weight} = 231\text{x}10^{-8}\left(\frac{120}{100}\right)^{1.8}(15.1)^4(1260)^2(4)^{.825}$$

$$= 832 \text{ lbs (This is within 3.2 percent)}$$

The formula appears to be reasonably accurate. Reference (12) indicates it to be within 3 percent accurate, but uses data which varies from that in Reference (13).

Rotor disk area, A, and blade tip speed, V_T, can be substituted for D and N, thus:

$$N^2D^2 = \left(\frac{60V_T}{\pi}\right)^2$$

$$D^2 = \frac{4A}{\pi}$$

Only shaft turbine engines will be used, so K = $.26\text{x}10^{-8}$.

$$W_R = .26\text{x}10^{-8}\left(\frac{AF}{100}\right)^{1.8}\frac{4A}{\pi}\left(\frac{60V_T}{\pi}\right)^2 B^{.825}$$

$$= .01184\left(\frac{AF}{100}\right)^{1.8}\left(\frac{V_T}{100}\right)^2 AB^{.825}$$

V_T = 800 is considered a good value for all ducted propellers

$$A = W_G/w$$

$$W_R = .01184\left(\frac{AF}{100}\right)^{1.8}\left(\frac{800}{100}\right)^2\frac{W_G}{w}B^{.825}$$

$$= .757\frac{W_G}{w}\left(\frac{AF}{100}\right)^{1.8}B^{.825}$$

Activity Factor, AF, is an expression for blade area/radius, in which greater weight is given for area near the blade tip.

B x AF corresponds approximately to rotor solidity. In propeller design it is increased as disk loading increases. As number of blades, B, can never be a fraction, AF is increased until its practical limit is reached. Then one or more blades are added and AF is abruptly reduced.

For purposes of this study it is desirable to have B and AF vary continuously. Therefore, AF has been held constant and B allowed to vary. This results in fractional blades, but is a method used for first approximations in actual propeller design.

AF was chosen = 100. This corresponds to conventional reciprocating engine propellers with disk loadings around 85 psf. This disk loading is near the center of the range considered in this report.

The aerodynamic section of Reference (14) gives the following expression for B x AF:

$$B \times AF = \frac{1360(1+f)w}{C_{L_R} V_T^2 \rho} \left(\frac{A_4}{A_2}\right)^2 \left[\frac{2.59}{\sqrt{1+\left(\frac{V_2}{V_T}\right)^2}} + \frac{1.0}{\sqrt{.09+\left(\frac{V_2}{V_T}\right)^2}} \right]$$

A_4/A_2 = The ratio of slip stream area downstream to that of the propeller.

= 1.0 for ducted propellers with straight exit ducting.

V_T = 800 fps

1+f = Flow area + equivalent flat plate area of drag surfaces.

= 1.3

C_{L_R} = Mean blade lift coefficient.

= .53 for optimum C_L/C_D.

V_2 = Down wash velocity.

$= \sqrt{\frac{w}{\rho}}$

AF = 100

$$B \times 100 = \frac{1360 \times 1.2w}{.53(800)^2 .001785}\left[\frac{2.59}{\sqrt{1+\frac{w}{.001785(800)^2}}} + \frac{1.0}{\sqrt{.09+\frac{w}{.001785(800)^2}}}\right]$$

$$B = .0269w\left[\frac{2.59}{\sqrt{1+\frac{w}{1142}}} + \frac{1.0}{\sqrt{.09+\frac{w}{1142}}}\right]$$

$$W_R = .757\,\frac{W_G}{w}\left(\frac{100}{100}\right)^{1.8}\left[.0269w\left(\frac{2.59}{\sqrt{1+\frac{w}{1142}}} + \frac{1.0}{\sqrt{.09+\frac{w}{1142}}}\right)\right]^{.825}$$

For a multi-rotation propeller the disk loading per hub w/H must be used and the entire term multiplied by the number of hubs per propeller, H.

$$W_R = .757\,\frac{HW_G}{w/H}\left[.0269\,\frac{w}{H}\left(\frac{2.59}{\sqrt{1+\frac{w/H}{1142}}} + \frac{1.0}{\sqrt{.09+\frac{w/H}{1142}}}\right)\right]^{.825}$$

$$W_R = .03835\,\frac{W_G}{w}\left[\frac{w}{H}\left(\sqrt{\frac{7650}{1142+w/H}} + \sqrt{\frac{1142}{102.7+w/H}}\right)\right]^{.825}$$

2.2 Transmission Weight, W_T

Reference (11) gives the following expression for the weight of a helicopter transmission:

$$W_T = .081\,Q^{.88}\left(\frac{n+1}{2}\right)^{.375}$$

Q = output shaft torque, ft lb

n = number of connecting shafts

In order to test its applicability, it is used on two transmissions of known weight and of the same type as those to be on the Flying Crane.

<u>Formula Test 1.</u> Allison T40 - A6, 5332 hp
14300 rpm input
15.7:1 gear ratio
2 input and 2 output shafts
Gear box weight: 803-822 lbs

$$Q = \frac{5250 \times \text{hp} \times \text{gear ratio}}{\text{input rpm}}$$

$$= \frac{5250 \times 5332 \times 15.7}{14300} = 30700 \text{ ft lb}$$

$$W_T = \left(\frac{2+1}{1}\right)^{.375} .081\ (30700)^{.88} = 876 \text{ lbs}$$

This is within 8 percent of the given weight, so the formula appears to be good.

(Note: It was found that the dual rotation output shafts must be considered as one shaft.)

<u>Formula Test 2.</u> Allison YT-56, 3017 hp
13820 rpm input
12.5:1 gear ratio
1 input and 1 output shaft
Gear box weight: 439 lbs

$$Q_{out} = \frac{5250 \times 3017 \times 12.5}{13820} = 14330 \text{ ft}$$

$$W_T = \left(\frac{1+1}{2}\right)^{.375} \times .081\ (14300)^{.88} = 402 \text{ lbs}$$

This is also within 8 percent of the given weight, so again the formula appears to be good.

It must now be put in terms of W_G and w .

$$Q = .250 \times hp/rpm$$

$$HP = .02376\ W_G \sqrt{w}$$

$$HP \text{ per duct} = .02376\ W_G/b \sqrt{w}$$

$$b = \text{no. ducts}$$

$$rpm = \frac{\text{rotor blade tip speed}}{\pi D}$$

Constant tip speed = 700 fps is used in all cases.

$$= 42000 \text{ fpm}$$

$$rpm = \frac{42000}{\pi D} \qquad D = \sqrt{\frac{4W_G}{\pi bw}}$$

$$= \frac{42000}{\pi} \sqrt{\frac{\pi bw}{4W_G}}$$

$$= 11860 \sqrt{\frac{bw}{W_G}}$$

$$Q = .5250 \times .02376 \frac{W_G \sqrt{w}}{b} \times \frac{1}{11860} \sqrt{\frac{W_G}{bw}}$$

$$= .01051 \left(\frac{W_G}{b}\right)^{3/2}$$

$$W_T = .0?1 \left(\frac{n+1}{2}\right)^{.375} \left[.01051\ \frac{W_G}{b}^{3/2}\right]^{.88} \quad \text{per duct}$$

$$W_T = .00147 \left(\frac{n+1}{2}\right)^{.375} \left(\frac{1}{b}\right)^{.32} W_G^{1.32} \quad \text{per ship}$$

2.3 Duct Weight, W

The only existing duct on which there was any available data was that on the Hiller 60" "Flying Platform", described in Reference (16). This only provided a duct weight for one set of conditions. As no well established theoretical information existed, it was necessary to construct a general theoretical expression for duct weight and then assign specific values to the expression by making it correspond to two known ducts. The 60" Flying Platform constituted one known duct and the other consisted of a "provisionally designed" duct 30 feet in diameter.

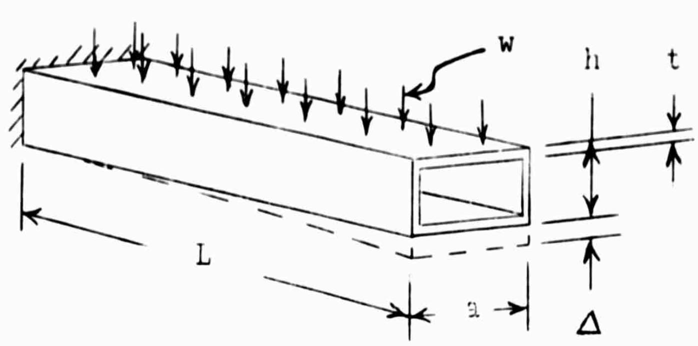
To form the general expression for duct weight it was considered that a duct must withstand structural conditions that are partly like those imposed on an aircraft wing and partly like those imposed on a fuselage.

An airplane wing can be considered as a cantilever beam with a uniform load, w lb per sq. ft of upper surface area. The beam has box type construction, with length, L, width, a, height, h, and wall thickness, t. The material of which it is made has density, ρ, and maximum working stress, S. The beam has constant external proportions. That is:

$$a = kL$$

$$h = k'L$$

t is small compared to a or h



Weight of beam

$$\text{Weight} = \rho L t\,(2h + 2a)$$

Moment at base of beam

$$M = \frac{SI}{C} = \frac{waL^2}{2}$$

$$\frac{I}{C} = \frac{2at(h/2)^2}{h/2}$$

$$= \frac{2kLt(k'L/2)^2}{k'L/2}$$

$$= kk'L^2t$$

$$kk'L^2tS = \frac{kwL^3}{2}$$

$$t = \frac{wL}{Sk'}$$

$$\text{Weight} = \rho L \times \frac{wL}{Sk'}\,(2k'L+2kL)$$

$$= \frac{2\rho}{Sk'}\,(k'+k) \times wL^3$$

Thus, for a beam (or wing) of given proportions and which is designed for bending strength, Weight $\sim wL^3$.

The weight of the same beam is now considered when it is designed for rigidity.

Deflection, Δ, must be in proportion to length, L. That is

$$\Delta = k''L$$

Also,

$$\Delta = \frac{waL^4}{8EI}$$

$$I = 2at(h/2)^2$$

$$= 2kLt(k'L/2)^2$$

$$= \frac{kk'^2tL^3}{2}$$

$$a = kL$$

$$\Delta = \frac{wkL \times L^4}{8Ekk'^2tL^{3/2}}$$

$$k''L = \frac{wL^2}{4k'^2Et}$$

$$t = \frac{wL}{4k'^2k''E}$$

$$\text{Weight} = \rho Lt(2h+2a)$$

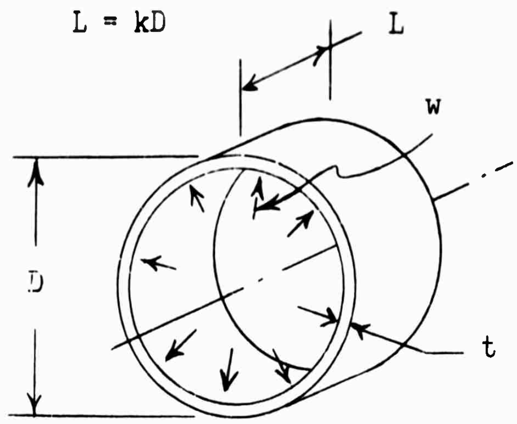
$$= \rho\,\frac{L \times wL}{4k'^2k''} \times 2L(k'+k)$$

$$= \rho\,\frac{(k+k')}{2k'^2k''} \times wL^3$$

Again, Weight $\sim wL^3$

A duct can also be considered as a cylindrical membrane loaded with a uniform pressure, w, and designed for bursting strength. It has constant proportions such that:

$L = kD$



Weight of cylinder $= \rho DLt$

$$= k\rho D^2 t$$

Bursting load, wLD = Bursting strength $2tLS$

$$t = \frac{wLD}{2LS}$$

$$= \frac{wD}{2S}$$

$$\text{Weight} = \frac{k\rho}{2S} \times wD^3$$

Again, weight $\sim w \times (\text{linear dimensions})^3$

The above three analogies indicate that a duct that is designed only to resist imposed aerodynamic loads will have weight proportional to wD^3. However, much of a duct is designed merely to support its own weight and to withstand accidental wear and tear. This is analogous to an airplane fuselage. Reference (15), Figure 38, gives airplane fuselage weight as:

$$\text{Fuselage Weight} = \text{constant} \times L\ (B + H)$$

L = length

B = width

H = height

Or, Weight $\sim (\text{linear dimensions})^2$

Reference (b) also shows fuselage weight as varying between airplanes of different speeds. The effect of speed on fuselage weight is shown below:

From Figure 38, for $L\ (B + H) = 1000\ f^2$

W = 2400 lb for V = 300 knots

W = 4900 lb for V = 500 knots

In general, it can be said that:

$$\frac{W_2}{W_1} = \left(\frac{V_2}{V_1}\right)^n$$

REVISION TO ERRATA

	Was	Change to
Page IV-10	$\ldots = \frac{wL}{Sk'}$	$\ldots = \frac{wL}{2Sk'}$
Page IV-10	Weight $= \rho L \times \frac{wL}{Sk}(2k'L + kL)$	Weight $= \rho L \times \frac{wL}{2Sk}(2k'L + 2kL)$
Page IV-[illegible]	$= \frac{2\rho}{Sk'}(k' + k) \times wL^{2}$	$\frac{\rho}{Sk'}(k' + k) \times wL^{2}$
Page IV-[illegible]	$\Delta = \frac{wkL \times L^{4}}{2Ekk'^{2} tL}{}^{1/2}$	$\Delta = \frac{wkL \times L^{4}}{4Ekk'^{2} tL}{}^{3}$
Page IV-1[illegible]	$= \rho \frac{L \times wL}{4k'^{2} k''} \times 2L(k' + k)$	$= \rho \frac{L \times wL}{4k'^{2} k''E} \times 2L(k' + k)$
Page IV-1[illegible]	$= \rho \frac{(k + k')}{2k'^{2} k''} \times wL^{3}$	$= \rho \frac{(k + k')}{2k'^{2} k''E} \times wL^{3}$
Page IV 22 (drawing)	$b = kL$	$a = kL$
Page IV 2[illegible]	Weight of beam $= \rho L \times (2a+b)t$	Weight of beam $= \rho L \times (2a+b)t$
Page IV 2[illegible]	$= \rho \frac{(SK + K')}{SKK'} \times PL^{2}$	$= \rho \frac{(2K + K')}{SKK'} \times PL$
Page IV 2[illegible]	Weight $\sim PL$	Weight $\sim PL$

REVISION TO ERRATA

Page II-5 Equation (17). $\sin\alpha = \frac{[illegible]\,\varepsilon}{2^2 + (1+\varepsilon)^2}$

Page II-6 Equation (18). $\cos\alpha = \frac{2V}{2^2 + (1+\varepsilon)^2}$

Page III-3
20,[illegible]00 lbs
37,900 lbs
96,000 lbs

Page V-4 Column headed $W_G \times 10^3$ should read $W_g \times 10^3$

Column headed W_P should read W_R

Column headed W_{TD} should read W_T

Page V-5 Column headed W_P should read W_R

Column headed W_{TD} should read W_T

ERRATA

Page II-[illegible] Equation (1[illegible]), $\sin c = \frac{[illegible]}{2\epsilon^2 - [illegible]\epsilon^2}$

Page II-6 Equation (1[illegible]), $\cos c = \frac{2[illegible]}{2\epsilon^2 - [illegible]\epsilon^2}$

Page III-3 20,200 lbs
37,900 lbs
96,000 lbs

Page V-[illegible] Column headed $W_G \times 10^3$ should read $W_E \times 10^{-3}$

Column headed W_P should read W_R

Column headed W_{TD} should read W_T

Page V-5 Column headed W_P should read W_R

Column headed W_{TD} should read W_T

Exponent, n, can be determined by substituting known values of W and V.

$$\frac{4800}{2400} = \left(\frac{500}{300}\right)^n$$

$$n = 1.4$$

This can be applied to duct loading by using the relation between speed and aerodynamic loading,

$$w = \frac{\rho V^2}{2}$$

$$w^{1/2} \sim V$$

$$\frac{W_2}{W_1} = \left(\frac{w_2}{w_1}\right)^{\frac{n}{2}}$$

$$= \frac{w_2}{w_1}^{.7}$$

A general expression for duct weight can now be stated.

$$W_D = kD^m w^n$$

Based on the wing and fuselage analogies,

$$m = 2 \text{ to } 3$$

$$n = 0 \text{ to } 1$$

The value of n = .7 derived above, is used.

k and m are determined by substitution of two sets of known values, with the limit, m = 2 to 3, used as a check.

Duct Weight Data

Case 1. Hiller Model 1031-A Flying Platform
Reference (16), page 13

Duct diameter (ID) = 60 inches

Weight = 22.0 lbs

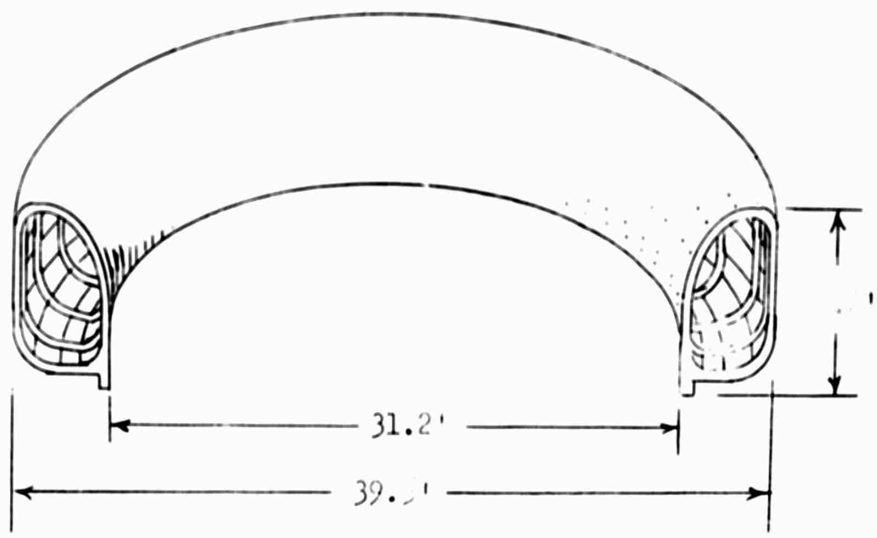
Disk loading = 12.5 psf

Note: The proportions of the Model 1031A duct and the ducts in this report are constant, namely:

$$OD = 1.27\ ID$$

$$Height = .287\ ID$$

Case 2. A duct of 31.2 feet ID with w = 35 psf is designed provisionally and its weight estimated. The duct is visualized as having conventional airplane fuselage type construction. (This would be conservatively heavy.)



The weight of an airplane fuselage of equivalent size can be obtained from Figure 36, Reference 15.

$$L = \pi \times \text{average diameter}$$

$$= \pi \frac{(31.2 + 39)}{2} = 110\ ft$$

$$H = 9\ ft$$

$$B = 4.2 \text{ ft}$$

$$L\,(B + H) = 110\,(9 + 4.2) = 1450 \text{ sq. ft}$$

For a 300 knot airplane this gives fuselage weight = 5000 lbs. Applying this data to the W_D formula,

$$W_D = kD^m w^{.7}$$

W_D	D	w
5000 lb	31.2 ft	35 psf
22	5.0	12.5

By trial and error k = .0542 and m = 2.6 are found to satisfy the above data.

$$W_D = .0542D^{2.6}w^{.7} \qquad \text{(per duct)}$$

Duct weight per ship, in terms of W_G and w

$$W_D = .0542b\sqrt{\left(\frac{4W_G}{\pi bw}\right)^{2.6}}\;w^{.7}$$

(per ship)

$$W_D = .103\left(\frac{W_G}{b}\right)^{1.3}\left(\frac{1}{w}\right)^{.6}$$

2.4 Engine Weight W_E

Reference (9), Chart I, shows the predicted weight of shaft turbine engines up to 1965. Interpolating between curves to 1962, and allowing for engines not being of the size that gives the lowest weight per power, it is estimated that engines will weigh .32 lb/hp. This is at standard sea level, with no ram effect, and ignoring lift obtained from downward jet exhaust.

ρ at 6000 ft and 95°F = 19.5/30.0 ρ at sea level.

Engine specific weight at 6000 ft and 95° = .32 x 30.0/19.5.

= .492 lb/hp

[illegible] W_F [illegible]

$$= [illegible] \times [illegible] \, W_G \sqrt{w}$$

$$[illegible] \, W_G \sqrt{w}$$

[illegible] W_{FA}

[illegible] starting system, and [illegible].

From Reference (11),

[illegible] W_{oil} [illegible] $(W_{F_{bare}})$ [illegible]

Starting System Weight, W_{start} [illegible] $(W_{F_{bare}})$ [illegible]

N = number of engines per ship

[illegible] is estimated to be 10% of $W_{E_{bare}}$.

As these weights [illegible] compared to W_G, they can be approximated by a simple expression.

$$W_{EA} = A \, W_{E_{bare}}{}^{n}$$

A and n are determined below.

W_{EA} for W_G [illegible]

w = 1[illegible]0 psf, 8 ducts

$$W_{oil} = 2.8 \times 8 + .049 \times 8^{.09} (9790)^{.906} = [illegible]0$$

$$W_{start} = .29 \times 8^{.40} (9790)^{.6[illegible]} = 209$$

$$W_{cowl} = \quad = 979$$

1468 lb

W_{EA} for W_G = 25000 lb,

w = 35 psf, 4 ducts:

$$W_{oil} = 3.8 \times 8 + .049 \times 8^{.09} (1730)^{.908} = 58$$

$$W_{start} = .29 \times 8^{.40} (1730)^{.60} = 73$$

$$W_{cowl} = \qquad = \frac{173}{325} \text{ lb}$$

W_{EA}	1468 lb	325 lb
$W_{E_{bare}}$	9790 lb	1730 lb

$$\left(\frac{1730}{9790}\right)^n = \frac{325}{1468}$$

$$n = .87$$

$$A = \frac{1468}{(9790)^{.87}} = .490$$

$$W_{EA} = .490\, W_{E_{bare}}^{\ .87}$$

$$= .490\, (.0117\, W_G \sqrt{w}\,)^{.87}$$

$$W_{EA} = .0102\, W_G^{\ .87}\, w^{.44} \qquad \text{(per ship)}$$

2.6. Structural Weight (Beams), W_{SB}

The method used in deriving an expression for structural weight was as follows:

(a) Overall sketchs of one 3-duct configuration and one 4-duct configuration were made to scale.

(b) Provisional designs were made of the structure for each configuration.

(c) The weight of each structure was calculated.

(d) A theoretical expression for structural weight was derived.

(e) Weight data calculated from the provisional designs was used to evaluate constants in the theoretical weight expression.

2.6.a. 3-Duct Configuration

$$w = 35 \text{ psf}$$

$$W_G = 80{,}000 \text{ lbs}$$

The first 3-view drawing shows the 3-duct configuration with $w = 35$ psf and $W_G = 80{,}000$ lbs. There are three engines per duct, driving counter-rotating propellers through transmissions located centrally in each duct. Each set of three engines is grouped around a pylon which extends downward to the landing gear. Each duct is connected to its engine nacelle by four spokes, and two of the interconnecting beams.

Figure 16 shows the arrangement of the spokes, beams, pylons, and power-plants. The upper end of each pylon is a ring to which are attached two beam upper longerons and four duct spokes. Six other spokes, arranged conically, extend downward to the landing gear strut.

Each engine is nested between two of the conically arranged spokes, and each transmission is in the center of a ring. The weights of the various aircraft components are distributed over the structure as is shown in Figure 17.

Provisional Beam Design for 3-Duct Configuration

The beams were designed to the following conditions:

Maximum vertical load (crash load) = 8.g ultimate.

Obstruction loads, applied at base of landing gear = 3.5g limit, vertical and 1.75g limit, horizontal in any direction.

In-flight load factors are expected to be less than those encountered in landing.

Margin of safety = 1.0

Construction to be of 2024ST Al Alloy tubing

F_{tu} = 64000 psi

F_{ty} = 42000 psi

To avoid local instability failure, tubing diameter/wall thickness was held = 30.

Each beam consists of an upper and lower tubular longeron, spaced 2.6 ft on centers with diagonal bracing of tubes set off at 45°. The tubing diameters were calculated to accommodate the loading conditions, the longeron cross-section area varying in uniform steps from one end of the beam to the other.

Cross-section properties of the longerons and diagonals were determined to be as shown in Figure 18.

Main Beam Weight, 3-Ducts

Specific weight, ρ, of 2024ST = .101 lb/in^3

Diagonals

$$\begin{aligned} \text{Weight} &= \rho V \\ &= .101\ LA = .101L\ A \\ &= .101 \times 2 \times \frac{2.6 \times 12}{\cos 45^\circ} \times 17.82 \\ &= 158.8 \text{ lbs} \end{aligned}$$

Longerons

$$\text{Average area} = \frac{13.61 + 3.40}{2} = 8.51 \text{ in}^2$$

Length = 2.6 x 12 x 4 x 8 = 998 in (Conservatively considering that beams extend to pylon centers.)

Weight = 8.51 x 998 x .101 = 857 lbs

Total Weight of Main Beam

Total weight = 857 + 158.8

= 1015.8 lb per beam

= 3048 lb per ship

2.6.b. Provisional Beam Design for 4-Duct Configuration

The second 3-view drawing shows the 4-duct configuration with W_g = 60,000 lb and w = 35 psf. There are two engines per duct, but otherwise the pylons and beams are of the same type as on the 3-duct configuration.

The weights of the various aircraft components are distributed on the structure, as is shown in Figure 19. The loading conditions are the same as were used for the 3-duct configuration, namely:

Crash loading = 8.g vertical, ultimate

Obstruction loading at base of landing gear = 3.5g vertical, limit and 1.75 horizontal, limit

Margin of safety = 1.0

Construction: 2024ST Al Alloy Tubing

F_{tu} = 64,000 psi

F_{ty} = 42,000 psi

D/t = 30

A provisional design shown in Figure 20 was made of the beams with the spacing between longerons and the cross section areas of members determined to accommodate the loads. The cross-section areas of members vary in uniform steps from one end of the beam to the other.

Provisional Design of Diagonal Beams, 4-Duct Configuration

See Figure 20

Longerons

$$\text{Average area} = \frac{11.65 + 7.00}{2} = 9.32 \text{ in}^2$$

$$\text{Length} = 47 \times 12 = 564 \text{ in}$$

Weight = (9.32) (564) (.101) = 530 lbs

Diagonals

Average area = 3.98 in^2

Length = 2.35 x 12 = 28.2 in

Weight = (20) (.101) (28.2) (3.98) = 226 lbs

Total Weight of Beam

Total weight = 2(530) + 226. = 1060. + 226. = 1286. lbs

Provisional Design of Outer Beams with Winch

See Figure 20

Longerons

$$\text{Average area} = \frac{3.70 + 14.60}{2} = 9.15 \text{ in}^2$$

Length = 36 x 12 = 432 in

Total weight = (9.15) (432) (.101) = 400 lbs

Diagonals

Average area = 3.45 in^2

Length = 2.55 x 12 = 30.6 in

Weight = (14) (.101) (30.6) (3.45) = 150 lbs

Total Weight of Beam

Total weight = 2(400) + 150 = 950 lbs

Provisional Design of Outer Beams without Winch

See Figure 20

Longerons

$$\text{Average area} = \frac{8.19 + 4.91}{2} = 6.55 \text{ in}^2$$

Length = 32 x 12 = 384 in

Weight = (6.55) (384) (.101) = 254 lbs

Diagonals

Average area = 2.72 in^2

Length = 2.30 in

Weight = (14) (2.30) (2.72) (.101) = 88 lbs

Total Weight of Beam

Weight = (2) (254) + 88 = 508 + 88 = 596 lbs

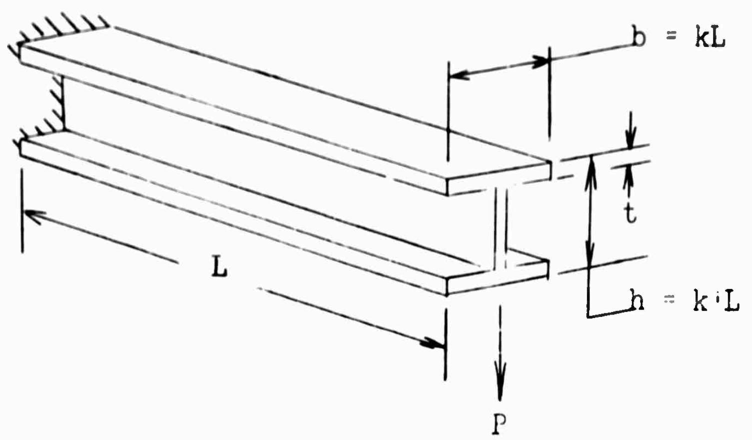
Total Weight of Beams of Ship

$$\text{Total weight} = 2\left[1286 + 950 + 596\right] = 2(2832) = 5660 \text{ lbs}$$

2.6.c. Beam Weight Equation

The main beams are basically considered to be simple beams with concentrated loads. The beam depth is proportional to length and the section area of the beam flanges is varied to accommodate the bending load.

The weight of such a beam illustrated is now investigated.



$$\text{Moment, } M = PL = \frac{SI}{C}$$

$$I = 2\,(at)\,\frac{h^2}{4}$$

$$= \frac{tK\;K'^2L^3}{2}$$

$$C = \frac{h}{2}$$

$$= \frac{k'L}{2}$$

$$PL = \frac{2}{K'L} \times StK\,\frac{K'^2L^3}{2}$$

$$PL = SK\;K'L^2t$$

$$t = \frac{P}{SKK'L}$$

$$\text{Weight of beam} = \rho L \times (2a+h)t$$

$$= \rho(2K+K')L^2 \times \frac{P}{SKK'L}$$

$$= \rho\,\frac{(SK+K')}{SKK'} \times PL^2$$

$$\text{Weight} \sim PL^2$$

In the case of the Flying Crane beams, P corresponds to W_G and L corresponds to duct diameter, D, which is proportional to $(W_G/w)^{1/2}$. In general,

$$W_{SB} = K \times W_G \left(\frac{W_G}{w}\right)^{1/2}$$

$$= K\,\frac{W_G^{3/2}}{w^{1/2}}$$

For 3-duct, 80,000 lbs, 35 psf, W_{SB} = 3048 lbs

$$K = \frac{W_{SB}\,w^{1/2}}{W_G^{3/2}}$$

$$= \frac{3048\ (35)^{1/2}}{(80,000)^{3/2}} = .000805$$

$$W_{SB} = \frac{.000805\ W_G^{\ 3/2}}{w^{1/2}} \qquad \text{per ship for 3-ducts}$$

For 4-duct, 80,000 lbs, 35 psf, $W_{SB} = 5660$ lbs

$$K = \frac{5660 \times .000805}{3048} = .001498$$

$$W_{SB} = \frac{.001498\ W_G^{\ 3/2}}{w^{1/2}} \qquad \text{per ship for 4-ducts}$$

2.6.d. Provisional Pylon Design

Using the pylon design and the loads from Section 2.6.a., a provisional design was made of a pylon for the 3-duct configuration. The cross-section areas of the pylon spokes, ring, oleo strut, and oleo piston, were determined. They are shown in Figure 21.

The pylon weight is calculated below:

Pylon weight, 3-ducts, W_G = 80,000 lbs, w = 35 psf (2024ST Al Alloy tubing construction)

Ring:

$$\text{Length} = \pi \times 4.5 \times 12 = 169.8''$$

$$\text{Average diameter} = \frac{13.0 + 12.12}{2} = 12.56''$$

$$\text{Wall thickness} = \frac{13.0 - 12.12}{2} = .44''$$

$$\text{Volume} = 169.8 \times 12.56\pi \times .44 = 2943 \text{ in}^3$$

$$\text{Weight} = .101 \times 2943 = 297 \text{ lbs}$$

Inner Spokes:

Total length = (6) x 5.5 x 12 = 396"

$$\text{Average diameter} = \frac{3.90 + 3.64}{2} = 3.77"$$

$$\text{Wall thickness} = \frac{3.90 - 3.64}{2} = .13"$$

$$\text{Volume} = 396 \times 3.77\pi \times .13$$

$$= 609 \text{ in}^3$$

Weight = 61.5 lbs

Oleo Cylinder:

Length = 8.7' x 12 = 104"

$$\text{Average diameter} = \frac{13.62 + 12.72}{2} = 13.17"$$

$$\text{Wall thickness} = \left(\frac{13.62 - 12.72}{2}\right) \times 2$$

$$= .90"$$

$$\text{Volume} = 104 \times 13.17\pi \times .90 = 3870 \text{ in}^3$$

Weight = 3870 x .101 = 392. lbs

Oleo Piston:

Length = travel + 3 diameters

= 16" + 36" = 52"

$$\text{Average diameter} = \frac{12.52 + 11.70}{2} = 12.11"$$

$$\text{Wall thickness} = \frac{12.52 - 11.72}{2} \times 2 = .80"$$

$$\text{Volume} = 52 \times 12.11\pi \times .80 = 1580 \text{ in}^3$$

Weight = 1580 x .101 = 160 lbs

Oleo Braces:

$$\text{Length} = 2 \times 8 \times 12 = 192''$$

$$\text{Average diameter} = \frac{5.20 + 4.85}{2} = 5.02''$$

$$\text{Wall thickness} = \frac{5.20 - 4.85}{2} = .18''$$

$$\text{Volume} = 192 \times 5.02\pi \times .18 = 545 \text{ in}^3$$

$$\text{Weight} = 545 \times .101 = 55 \text{ lbs}$$

Pylon tubing total weight = 55.0
160.0
245.0
61.5
297.0
718.5 lbs

+ 10% for fittings = 72.0
+ Foot = 250.0
1040.0 (Total weight of one pylon, for external loading, 3 ducts)

W_{SP} = 3120 lbs

2.6.e. Derivation of General Equation for Pylon Weight

A general equation for pylon weight is now derived. To do this, the manner in which pylon height varies with W_G must be investigated. Figure 20 shows the parts of pylon length for a ship of W_G = 64000 lbs and w = 35 psf.

Minimum ground clearance = duct diameter x sin 2°. Duct diameter varies with $W_G^{1/2}$. Beam depth = 2.5' for W_G = 65000 lbs. Beam depth varies with $W_G^{1/3}$.

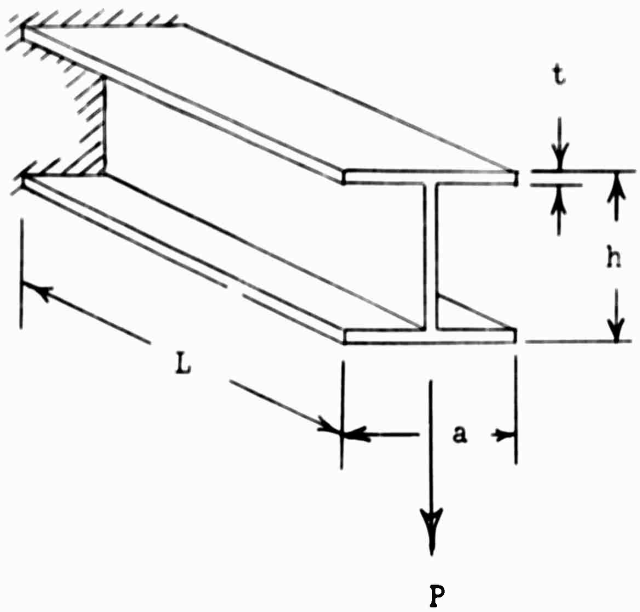
$$\text{Pylon height} = 11.5 + 2.5 \times \frac{W_G}{64000}^{1/3} + .5' \times \frac{W_G}{64000}^{1/2}$$

Ht	14.5'	15.36
W_G	64000	128000

$$\frac{128000^n}{64000} = \frac{15.36}{14.5}$$

$$n = .084$$

The pylon is likened to a beam whose length $\sim$ [illegible]$^{.084}$ and whose section properties are varied to accommodate the bending load. Its section has constant proportions, as shown in the sketch below:



$$M = PL = \frac{SI}{C}$$

$$L = k\,P^{.084}$$

$$I = 2\,at\,(h/2)^2$$

$$a = k'h$$

$$t = k''h$$

$$I = \frac{k'k''h^4}{2}$$

$$C = \frac{h}{2}$$

$$PL = k\,P^{1.084}$$

$$\frac{SI}{C} = S\,k'k''h^3$$

$$h = \left(\frac{k\,P^{1.084}}{S\,k'k''}\right)^{1/3}$$

$$h \sim P^{.362}$$

$$\text{Weight} \sim Lh^2$$

$$\sim P^{.084\,+\,2\,\times\,.362}$$

$$\sim P^{.808}$$

<u>General Expression for Pylon Weight</u>

By analogy with simple beam

$$W_{SP} = K\,W_G^{.808}$$

$$K = \frac{W_{SP}}{W_G^{.808}}$$

For 3-ducts, W_G = 80,000 lbs, external load, W_{SP} = 3120 lbs

$$K = .165 \times \frac{3120}{[illegible]} = .342$$

For 4-ducts,

$$W_{SP} = \frac{4}{3} \times \left(\frac{3}{4}\right)^{.808} \times W_{SP} \text{ for 3-ducts}$$

$$= 1.06 \times W_{SP} \text{ for 3-ducts}$$

$$= 1.06 \times .342$$

$$= .361$$

$$W_{SP} = .342 W_G^{.808} \text{ (for 3-ducts)}$$

$$W_{SP} = .361 W_G^{.808} \text{ (for 4-ducts)}$$

2.7 Structural Accessories Weight, W_{SA}

Structural accessories consist of flight controls, hydraulic and electric systems, furnishings, and [illegible]. Reference 11, page [illegible], gives the following expressions for the first three:

$$W_{controls} = .512\, W_G^{.68}$$

$$W_{hyd\ and\ elec} = .381\, W_G^{.71}$$

$$W_{furnish} = .682\, W_G^{.5} - 50$$

These three expressions can be replaced by one expression of the form,

$$W_{SA} = K\, W_G^{\,n} - 50$$

K and n are evaluated below:

$$\left(\frac{W_G'}{W_G''}\right)^n = \frac{W'_{con} + W'_{hyd} + W'_{furn} + 50}{W''_{con} + W''_{hyd} + W''_{furn} + 50}$$

$$\left(\frac{50000}{200000}\right)^{n} = \frac{3579}{6727}$$

$$n = .4[illegible]$$

Cabin weight is estimated to be [illegible]00 [illegible]s for W_G = [illegible]0 lbs, [illegible] as follows:

$$W_{cabin} = .[illegible] W_G^{.[illegible]}$$

$$K (50000)^{.[illegible]} = 50[illegible] [illegible]9 [illegible] 50[illegible] [illegible] 9 [illegible])^{.57}$$

$$K = [illegible] 1$$

$$W_{SA} = [illegible] W_G^{.7} - [illegible]$$

3.[illegible] [illegible] W[illegible] s. W

W [illegible] s [illegible] e[illegible] e[illegible], W[illegible]s[illegible] + Ra[illegible] weight, W[illegible].

Reference [illegible], [illegible] s [illegible]:

$$W_{irs} = .[illegible] W_G$$

Communications equipment weight has been determined [illegible] in [illegible] study as,

$$W_{radio} = [illegible]73 \text{ lbs}$$

$$W = [illegible]73 + .[illegible]10[illegible] W_G^{.71} \quad \text{(per ship)}$$

3. FUEL AVAILABLE

The ratio of fuel available to gross weight is given by the following expression:

$$R_{F_{avail}} = K \left(1 - \frac{W_P}{W_G} - \emptyset\right)$$

The empty weight ratio $\emptyset$ is the empty weight divided by gross weight. Empty weight is the sum of the various component weights [illegible] ssi[illegible]s of Part 2. The first [illegible] of the empty weight ratio is [illegible] e[illegible] i[illegible] e[illegible].

The constant K is the ratio of fuel weight to fuel weight plus tank weight. It is assumed that the jet fuel weight is 6.5 pounds per gallon and the tank weight is 0.5 pounds per gallon.

Hence, $$K = \frac{6.5}{6.5 + .5} = .928$$

The payload ratio $\frac{W_P}{W_G}$ is the ratio of weight of payload plus crew to gross weight. W_P is the weight of cargo and crew. The crew is assumed to weigh 600 pounds.

Hence, $$W_P = W_{cargo} + 600 = P + 600.$$

Ratio of fuel available to gross weight is presented in the following final form:

$$R_{F_{avail}} = 0.928 \left(1 - \frac{P + 600}{W_G} - \psi\right)$$

FLYING CRANE (DUCTED FAN PHASE) - DETERMINATION OF FUEL AVAILABLE

TABLE I - EMPTY WEIGHT EXPRESSION, "ϕ"

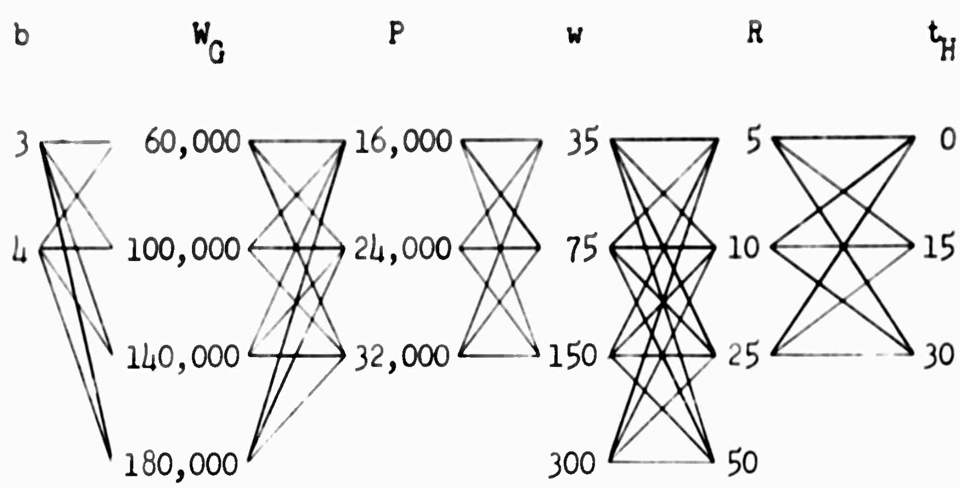
	ROTORS	TRANSMISSIONS	ENGINES		STRUCTURES				OTHER
	W_R	W_T	W_E	W_{EA}	W_{SB}	W_{SP}	W_D	W_{SA}	W_O
$\phi = \frac{1}{W_G} \times$	$.03835 \frac{HW_G}{w^{.175}} \left(\sqrt{\frac{7650}{1142H+w}} + \sqrt{\frac{1142}{102.7H+w}} \right)^{.825}$ H = 1 for w $\leq$ 10 psf H = 2 for w > 10 psf	+ $.00147 \left(\frac{n+1}{2} \right)^{.375} \frac{W_G^{1.32}}{b^{.32}}$ b = number of ducts n = 3 for w $\leq$ 10 psf and b = 4 n = 4 for w > 10 psf and b = 4 n = 4 for w $\leq$ 10 psf and b = 3 n = 5 for w > 10 psf and b = 3	+ $.0117 W_G \sqrt{w}$	+ $.0101 W_G^{.87} w^{.44}$	+ $\frac{kW_G^{3/2}}{w^{1/2}}$ k = .000805 for 3 ducts k = .001498 for 3 ducts	+ For Internal Loading $kW_G^{2/3}$ k = 5.32 for 3 ducts k = 5.84 for 4 ducts For External Loading $k(W_G - W_P)^{2/3}$ k = 2.04 for 3 ducts k = 2.23 for 4 ducts	+ $\frac{.103 W_G^{1.3}}{W_D^{.6} b^{.3}}$	+ $3.962 W_G^{.67} - 50$	+ $273 + .104 W_G^{.71}$

V. PARAMETRIC STUDIES

1. BASIC PARAMETERS

The basic parameters in this study are number of ducts b, gross weight W_G, payload P, disk loading w, mission radius R, and hover time t_H. Radius and hover time are not included in the parameters of fuel available study. Number of ducts is eliminated in fuel required study by two assumptions. One, the specific fuel consumption is assumed to be a function of total horsepower rather than a sum of functions of individual engine horsepowers. Secondly, the power-correction factor k_c that takes into account the effect of control moments is assumed to be 1.04 for both configurations.

The values of parameters that are used in fuel to weight computations are written in the following matrix form.



2. RESULTS OF FUEL TO WEIGHT RATIO COMPUTATIONS

Required R_F

With other parameters constant, R_F increases with increasing radius, disk loading, and hover time. This is as expected. Similarly, R_F decreases slightly with increasing payload. The decrease in horsepower required in return flight with increasing payload is responsible for this change. These trends of the variations of parameters on the R_F-W plot are presented in Figures 22, 23, 24, and 25. The figures also show, that with all these parameters constant, R_F increases with increasing gross weight. The increasing ratio of horsepower required to gross weight with increasing gross weight in return flight produces the variation.

Available R_F

Available R_F is affected mainly by gross weight, payload, and disk loading, see Figures 26 and 27. The curves show that R_F increases with design gross weight W_G, leveling off in the region where W_G is very large compared to payload.

Figure 26 indicates that up to a certain disk loading, R_F increases with w. Above that value of w, R_F decreases again. This is due to the large weights of structure, ducts, and rotors for low disk loadings. On the other hand, at high disk loadings, w, the engine weight is considerable. For any given design gross weight, the empty weight has a minimum in the region of w = 150 lb/ft^2.

The increase in weight of ducts and connecting structure is slightly more than the corresponding increase in the design gross weight. This tends to penalize large size so that at very large values of W_G, R_F actually decreases. Likewise, for a given W_G the 4-duct configuration, having smaller ducts than the 3-duct configuration, will have a lower duct weight. However, it will have a larger structural weight, so the net difference is small. See Figure 28.

3. RESULTS OF OPTIMIZATION STUDY

The required and available fuel to weight ratios are plotted for all combinations of payload, disk loading, radius, hover time, and duct configuration. The intersections of the required and available R_F curves give the possible combinations of gross weight and disk loading to carry out the mission. The optimum ship for the mission is the ship of least gross weight. Figure 29 is a typical intersection plot which illustrates the method of obtaining the optimum ship. Trends of variations of payload, hover time, and radius on gross weight, disk loading, and fuel to weight ratio are presented in Figures 30, 31, and 32. As shown by the graphs, with other parameters constant, the gross weight increases with increasing payload, hover time, and radius, the optimum disk loading decreases with increasing hover time and radius and increases with increasing payload; the fuel to weight ratio increases with increasing payload, hover time, and radius. The difference between 3 and 4 duct configurations of the optimum ships is small in general. The largest difference lies in the region of small radius and large payload

[illegible]

[illegible] exceed [illegible] A list [illegible] it is believed that the data shown [illegible] Also [illegible] included in the [illegible] upon the [illegible] that the [illegible]

TABLE II - LIST OF OPTIMUM SHIPS FOR 3-DUCT CONFIGURATION

P Lbs	Range Naut. Miles	t_H Min.	R_F	W_Gx10 ^{3} Lbs	w Lb./ft ^{2}	%TNRP Fwd. Flt.	%TNRP Ret. Flt.	%TNRP Avg.	W_E	W_{EA}	W_{SB}	W_{SP}	W_D	W_{SA}	W_O	W_P	W_{TD}
16,000	10	0	.043	36.3	134	74.5	31.5	53.0	4800	820	510	1640	3500	2470	452	2750	1400
		15	.072	38.9	111												
		30	.103	43.7	88												
	20	0	.062	38.3	116												
		15	.085	42.0	94												
		30	.115	46.4	80												
	50	0	.110	44.2	92												
		15	.135	50.0	74	76.4	45.2	60.8	5000	920	1020	2120	9200	3070	498	4100	2120
		30	.157	56.8	66												
	100	0	.190	62.4	75												
		15	.207	73.8	67												
		30	.230	90.0	59												
24,000	10	0	.042	54.0	178												
		15	.073	60.6	116												
		30	.105	67.3	88												
	20	0	.065	58.0	136												
		15	.095	64.5	101												
		30	.118	71.1	80	75.8	42.7	59.3	7500	117	1650	2840	10800	3890	562	5700	3390
	50	0	.107	67.8	85												
		15	.140	76.6	81												
		30	.166	88.8	74												
	100	0	.193	101.0	75												
		15	.220	130.4	72												
		30	.255	180.4	70												
32,000	10	0	.040	71.3	197												
		15	.072	79.9	116												
		30	.109	91.0	100												
	20	0	.068	77.0	161												
		15	.098	86.7	105												
		30	.127	98.0	92												
	50	0	.118	93.3	104												
		15	.147	107.0	90												
		30	.176	126.0	82												
	100	0	.210	156.5	86	75.5	54.5	65.0	19,200	255	4950	5390	25800	6650	781	12200	9550
		15															
		30															

TABLE II - LIST OF OPTIMUM SHIPS FOR 4-DUCT CONFIGURATION

P Lbs	Range Naut. Miles	t_H Min.	R_F	$W_G x10^{-3}$ Lbs	w lb/ft^2	%TNRP Fwd. Flt.	%TNRP Ret. Flt.	%TNRP Avg.	W_E	W_{EA}	W_{SB}	W_{SP}	W_D	W_{SA}	W_O	W_P	W_{TD}
16,000	10	0	.040	35.8	150	74.5	30.5	52.5	5100	850	800	1710	2800	2440	450	2450	1100
		15	.073	39.1	108	74.7	34.9	54.8	4600	760	1180	1850	4000	2600	462	2950	1220
		30	.103	43.6	91	75.2	39.1	57.2	4900	800	1480	2020	5000	2790	478	3440	1430
	20	0	.059	38.2	112	74.5	33.9	54.2	4650	790	1090	1820	3700	2560	459	2830	1200
		15	.090	42.0	98	75.0	37.4	56.2	4900	790	1380	1950	4600	2720	472	3220	1360
		30	.120	46.3	84	75.6	41.2	58.4	5000	810	1650	2110	5600	2900	487	3720	1550
	50	0	.105	43.8	88	75.4	39.8	57.6	4750	790	1500	2025	5100	2800	478	3450	1420
		15	.140	50.0	79	75.7	44.0	59.9	5300	850	1880	2250	6200	3060	499	4000	1700
		30	.162	56.3	72	76.6	48.7	62.7	5600	880	2380	2470	8200	3320	519	4680	2000
	100	0	.194	62.6	79	76.0	50.1	63.1	6500	1040	2650	2700	8600	3570	538	5050	2300
		15	.214	74.2	72	76.6	54.8	65.7	7300	1140	3670	3100	11500	4000	572	6190	2900
		30	.245	91.9	65	77.5	59.7	68.6	8500	1290	5400	3690	16500	4630	620	7850	3840
24,000	10	0	.042	53.0	175	74.5	30.8	52.7	8100	1250	1400	2360	4300	3180	508	3490	1840
		15	.075	60.0	102	74.8	36.0	55.4	7000	1090	2300	2600	7200	3470	530	4580	2180
		30	.112	67.0	95	75.0	40.3	57.7	7600	1160	2810	2810	8650	3740	551	5260	2530
	20	0	.060	56.8	110	74.6	33.8	54.2	6800	1080	2000	2490	6350	3340	520	4260	2010
		15	.091	64.0	95	75.1	38.7	56.9	7250	1100	2620	2750	8200	3620	542	5000	2390
		30	.125	72.2	84	75.6	43.1	59.4	7700	1200	3250	3025	10100	3930	566	5810	2800
	50	0	.107	68.7	85	75.5	41.5	58.5	7400	1140	3000	2900	9450	3790	555	5500	2600
		15	.140	77.7	81	75.7	45.2	60.5	8100	1260	3670	3210	11250	4125	581	6280	3080
		30	.162	88.7	70	76.8	50.0	63.4	8600	1300	4830	3580	14700	4520	612	7420	3670
	100	0	.182	99.7	70	76.8	52.8	64.8	9700	1430	5780	3950	17150	4890	640	8350	4280
		15	.220	132.8	70	76.8	58.6	67.7	12900	1610	8900	4980	24800	5950	725	11150	6240
		30	.260	216.0	65	77.5	66.5	72.0	20200	2830	18300	7500	47400	8460	924	18500	11480
32,000	10	0	.041	72.0	175	74.5	31.8	53.2	11000	1620	2250	3020	6500	3920	565	4740	2790
		15	.076	80.3	120	74.5	36.1	55.3	10200	1500	3300	3310	9700	4220	589	5930	3210
		30	.111	90.4	106	74.7	39.9	57.3	10650	1490	4150	3650	12000	4575	617	6900	3760
	20	0	.071	77.8	145	74.5	34.6	54.6	10900	1620	2750	3220	7850	4130	581	5410	3080
		15	.099	86.0	108	74.6	38.4	56.5	10350	1520	3800	3500	11100	4430	604	6520	3520
		30	.133	98.5	99	74.9	42.9	58.9	11350	1650	4860	3910	14000	4860	638	7650	4200
	50	0	.121	96.1	111	74.5	41.8	58.2	11800	1710	4450	3850	12700	4780	632	7250	4090
		15	.153	108.5	90	75.3	46.0	60.7	11900	1740	5850	4240	16600	5190	665	8590	4790
		30	.178	130.6	86	75.5	50.9	63.2	14100	2010	7850	4910	21600	5880	720	10450	6100
	100	0	.210	164.2	83	75.6	55.7	65.7	17200	2400	11100	5900	29400	6870	800	13100	8200
		15	.249	208.0	82	75.7	59.8	67.8	21800	3050	15180	7250	39500	8200	902	16650	11000

VI. CONCLUSIONS

Contrary to the power required curve of the helicopter, which has a pronounced minimum at a forward speed corresponding to a tip speed ratio of approximately 0.[illegible], the power required for a ducted propeller is much less dependent of speed. Broadly speaking, for the same weight and disk loading and for the speed range investigated, the power required for a ducted propeller configuration is about the same as that of the helicopter at its speed of minimum power. This increase in efficiency is, of course, due to the beneficial effect of the shroud.

However, if the shroud is laid out for maximum efficiency in hovering, relatively large nose-up pitching moments are generated in forward flight. These pitching moments, which have to be compensated by proper means of control, can represent a serious problem and should, therefore, not be overlooked. The rather limited test data presently available indicate that these pitching moments can be reduced considerably by using a shroud form, which is less advantageous in hovering. This means that, unless a better type of shroud can be developed, the designer has to compromise between hovering efficiency and forward flight characteristics.

If no additional means of propulsion are used, i.e., if the total drag is overcome by the horizontal component of the thrust vector(s), large forward tilt angles are required in forward flight. To a certain extent, deflection of the slip-stream by vanes has the same effect as tilting of the ducts. As tilt angles up to 50° are needed, tilting of the whole fuselage is practically out of the question. This means that, unless additional means of propulsion are employed, either the ducts have to be tilted relative to the fuselage or that the principle of slip-stream deflection has to be used. The possible range of application and efficiency of the latter is not yet fully known.

The other possibility is to employ separate means of propulsion such as additional propellers. To the extreme, the aircraft would have essentially zero angle of incidence at all level flight conditions, where the total drag is overcome by the additional propellers. The momentum theory indicates that in this case the total power required for forward flight increases considerably. According to Hiller's truck tests of the flying platform the nose-up pitching moments also increase.

For these reasons, this contractor presently favors a compromise consisting of slip-stream deflection in connection with a propulsive propeller and slight forward tilt of the aircraft in forward flight. Whether such an additional propeller is required or not depends to a high extent on both the efficiency of the slip-stream deflection method and on the stalling characteristics of a ducted fan in forward flight.

The answer to these problems is not known at present. It is felt, therefore, that the final answer on the best method of propulsion must be obtained from wind tunnel tests.

The missions investigated in this report cover the following ranges:

Radius:	5 to 50 nautical miles
Hover time:	0 to 30 minutes
Payload:	16,000 to 32,000 pounds

With regard to the optimum ships found by the parametric studies, the following statements can be made. Gross weight, disc loading, and fuel to weight ratio for a given mission are affected only slightly by a change in number of ducts from 3 to 4 per ship. In general, an increase in the duct number from 3 to 4 decreases the optimum disc loading and increases the minimum gross weight. The fuel to weight ratio remains practically the same.

The effects of the variables of mission (radius, hover time, and payload) upon the design parameters of the ship (gross weight, disk loading, and fuel to weight ratio) may also be expressed in general trends. As expected, gross weight increases with increasing payload, hover time and radius. Optimum disc loading decreases with increasing hover time and radius, and increases with increasing payload. The fuel to weight ratio increases with increasing payload, hover time, and radius.

A detailed list of the parameters of the optimized aircraft is given in Section V,3 of this report. Broadly speaking, the resulting parameters lie within the following limits:

Gross weight:	36,000 to 208,000 lbs
Disc loading:	60 to 197 lb/ft^2
Fuel to weight ratio:	0.04 to 0.260

As pointed out previously, due to the lack of basic information the above results are partly based on theoretical performance calculations derived from the momentum theory. Interference effects have been neglected. In the course of the study various other assumptions had to be made which are described in the body of this report.

Although it is believed that these assumptions are realistic and that the theoretical developments represent the present state of the art, it should always be borne in mind that for the final design of a successful ducted fan type flying crane further experiments are required to give the answer to the problems still unsolved today.

VII. REFERENCES

1. G. Clancy, R. Cowgill: Truck Test-Stand Tests of Hiller Airborne Personnel Platform, Hiller Engineering Report 680.2, September, 1955.

2. R. J. Platt, Jr.: Static Tests of a Shrouded and an Unshrouded Propeller, NACA Report RML7H25, 1948.

3. W. Krueger: On Wind Tunnel Tests and Computations Concerning the Problem of Shrouded Propellers, NACA TM 1202, 1949.

4. H. H. Hubbard: Sound Measurements for Five Shrouded Propellers at Static Conditions, NACA TN 2024, 1950.

5. A. Stone: A Study of Shrouded vs Unshrouded Propellers, BuAer Report DR 1750, 1955.

6. L. P. Parlett: Aerodynamic Characteristics of a Small-Scale Shrouded Propeller at Angles of Attack from 0 to 90°, NACA TN 3547, 1955.

7. W. T. Grady, Wright Air Development Center, Wright-Patterson Air Force Base: Some Aerodynamic Aspects of Propellers for VTOL Aircraft, Paper Presented at Western Forum, American Helicopter Society, Dallas, Texas, October, 1956.

8. N. E. Nelson, Doak Aircraft Company, Inc.: The Advantages of the Ducted Propeller in VTOL Aircraft Design, Paper Presented at Western Forum, American Helicopter Society, Dallas, Texas, October, 1956.

9. R. Booth: Generalized Shaft Turbine Engine Characteristics, Hiller Engineering Report 630.5, December, 1955.

10. D. Joy: The R_F Graphical Method of Parametric Analysis for the Development of Optimum Preliminary Design Aircraft, Advanced Research Division of Hiller Helicopters Report 107, February, 1956.

11. D. P. Joy and R. M. Simonds: Transport Helicopter Design Analysis Methods, Hiller Engineering Division Report No. 473.6, January 30, 1955.

12. R. K. Tiedeman: Estimating Formula for Basic Propeller Weight, Curtiss-Wright Corp., Propeller Division, letter of January 12, 1954.

13. Catalog of Propellers, Curtiss-Wright Corp., Propeller Division, September 1954.

14. (Being prepared concurrently with this report): Summary Report of Airborne Personnel Platform, Phase II, Hiller Engineering Division.

15. Aircraft Design Analysis Methods as Employed by the Research Division of the Bureau of Aeronautics, U.S. Navy Department, U.S. Navy NavAer D.R. Report No. 1139, October 6, 1949.

16. J. E. Johnson: Actual Weight Report, Flying Platform (Gear Driven), Hiller Engineering Report No. 310.54, April 5, 1956.

VIII. LIST OF SYMBOLS

W weight in general, lbs

W_G design gross weight, lbs

W_o actual weight at the beginning of a cruise or hover period, lbs

D propeller diameter, ft

A propeller disk area, ft^2

A_e duct exit area, ft^2

b number of ducts

w nominal propeller disk loading, lb/ft^2, parameter of parametric study

$$w = \frac{W}{bA}$$

w_e true loading of duct exit area, lb/ft^2, used for forward flight performance calculations and assumed to be

$$w_e = 1.1w$$

T propeller force, lbs
for hovering: net thrust per propeller-duct combination
for forward flight: resultant force vector, see Figure 4

M figure of merit for static thrust, see equation (1)

τ efficiency factor for forward flight, defined similar to M, see equations (24)(25)

V_o flight velocity, ft/sec

V_e duct exit velocity, ft/sec

V_c vertical rate of climb, ft/sec

γ angle of climb, deg

ε velocity ratio, $\varepsilon = V_e/V_o$

α duct tilt angle, deg, positive if duct is tilted forward

D_e external drag, lbs

D_i internal drag, lbs

f_e non-dimensional parameter for external drag, defined by equation (14)

f_i non-dimensional parameter for internal drag, defined by equation ([illegible])

$\dot{m}$ mass flow per second, lb sec/ft

$\dot{m} = \rho V_e A_e$

F function defined as

$F = \varepsilon(\varepsilon^2 - 1)$, see also Figure 9

F' $F' = \frac{F}{\Gamma}$ see equations (27)(28) and Figure 10

Γ non-dimensional lift parameter, see also Figure 5

$\Gamma = \frac{w_o}{V_o^2 \rho}$

C_m pitching moment coefficient, positive nose-up, defined as

$C_m = \frac{\text{Pitching Moment}}{A \rho D}$

$\zeta, \Delta\zeta$ non-dimensional coefficients for power required, see equation (32); ζ refers to power required in level flight, $\Delta\zeta$ to excess power available for climb

$\Delta\zeta/\zeta$ ratio (excess power available for climb)/(power required for level flight)

η_c climbing efficiency, see equations ([illegible])([illegible])

η_p propeller efficiency

η_t transmission efficiency

η overall efficiency

$\eta = \eta_p \eta_t$

k_c performance correction factor that takes into account the increase in power required due to pitch control (differential thrust), assumed to be 1.04 at 70 knots

ρ density of air, lb $sec^2 ft^{-4}$

q dynamic pressure, lb/ft^2

$$q = V_o^2 \rho/2$$

HP horsepower

R range, nautical miles

SFC specific fuel consumption, lbs/HP/hr

In addition, the following symbols have been used for Section IV "Derivation of Weight Equations":

H number of hubs per duct (= 1 for single rotation, 2 for dual)

k constant. Used for several different cases

n number of input shafts per transmission

R_F fuel weight/gross weight

W_G gross weight of ship

W_R rotor weight

W_T transmission weight

W_D duct weight

W_E engine weight

W_{SB} structure weight (pylons)

W_{EA} engine accessory weight (oil + tanks + starting system)

W_{SA} structural accessory weight (electric and hydraulic systems, cabin, furnishings, and flight controls)

W_O "other" weight (instruments and radio)

W_P payload (cargo and crew)

W_C cargo weight

FIGURE 2

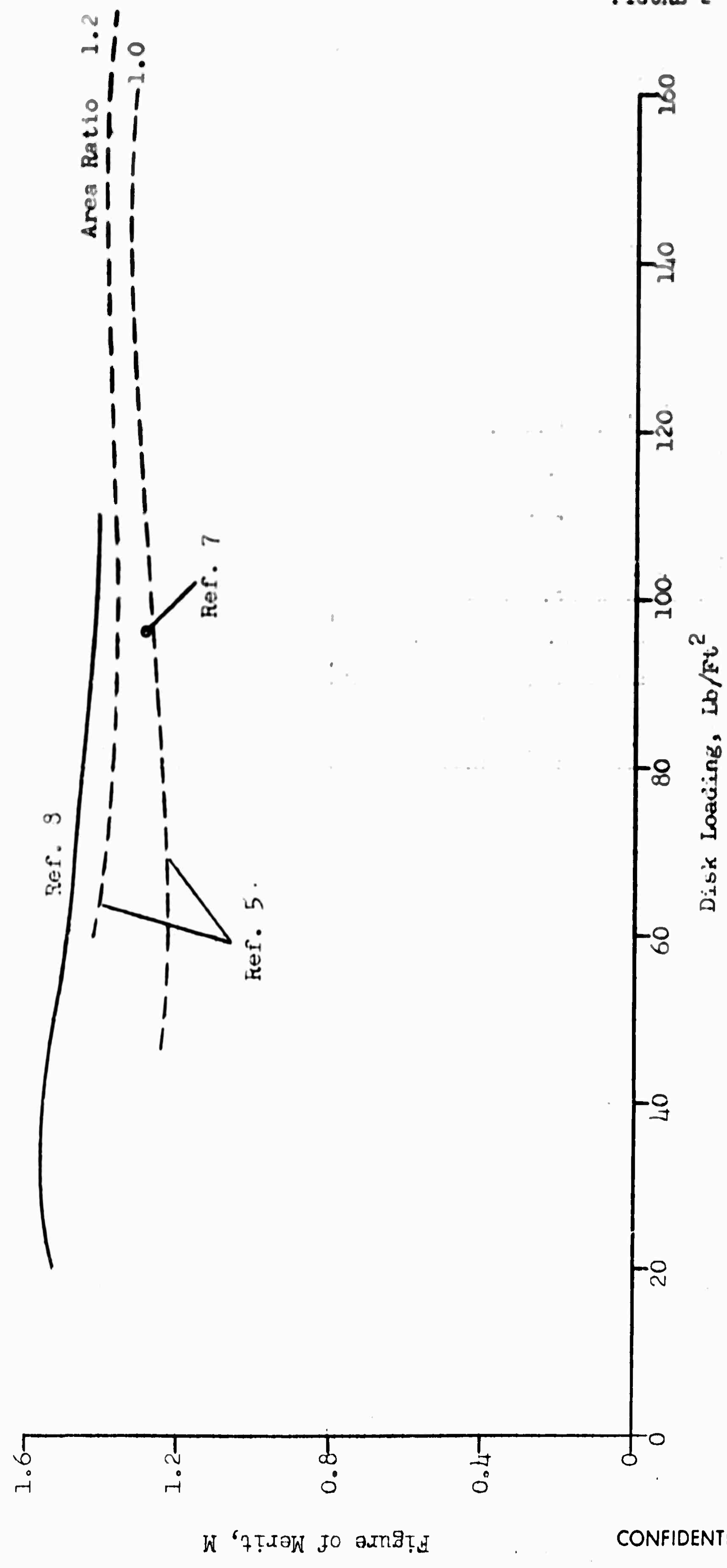


FIGURE 2: FIGURE OF MERIT VS. DISK LOADING

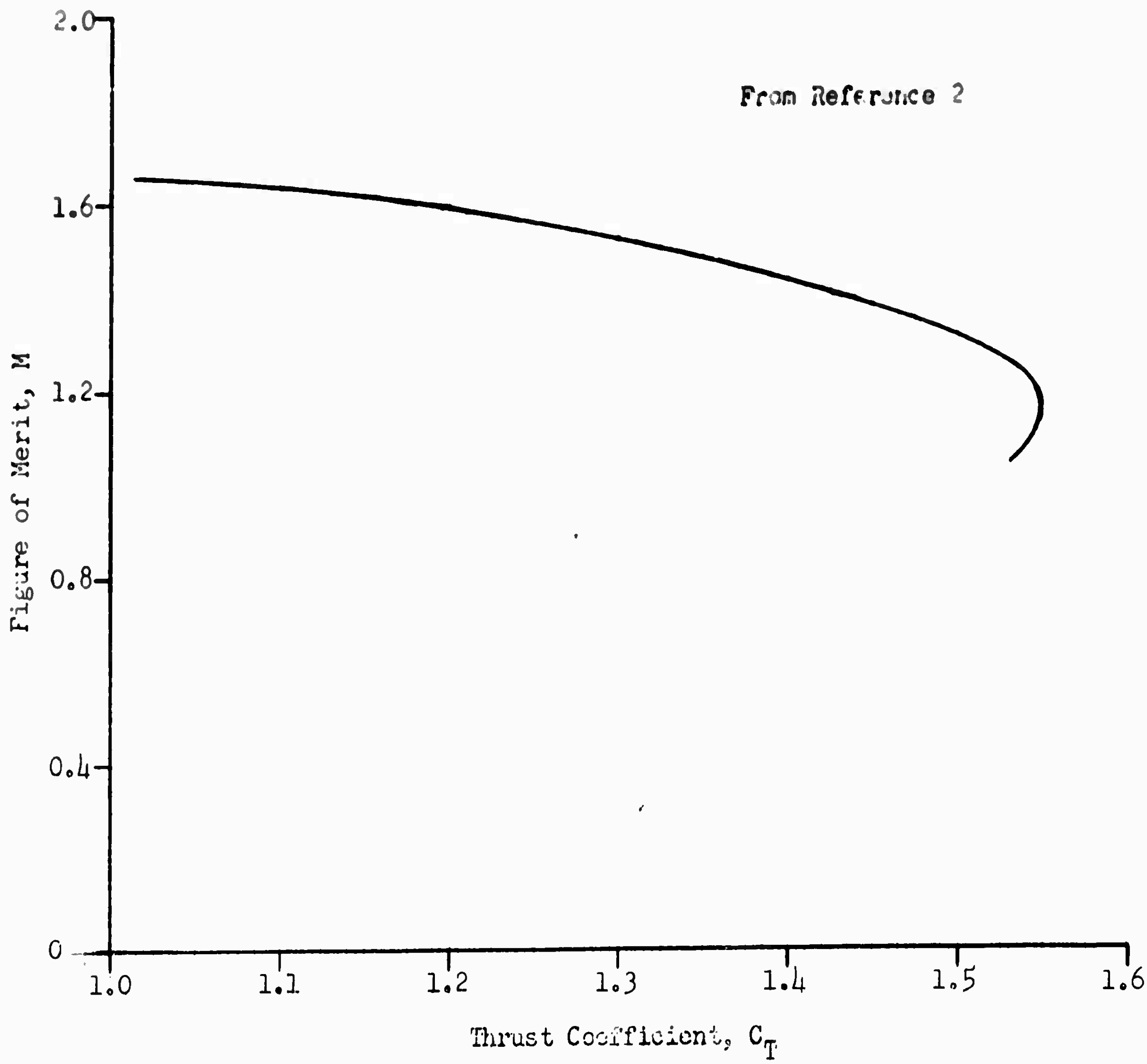


FIGURE 3: FIGURE OF MERIT OF SHROUDED PROPELLER

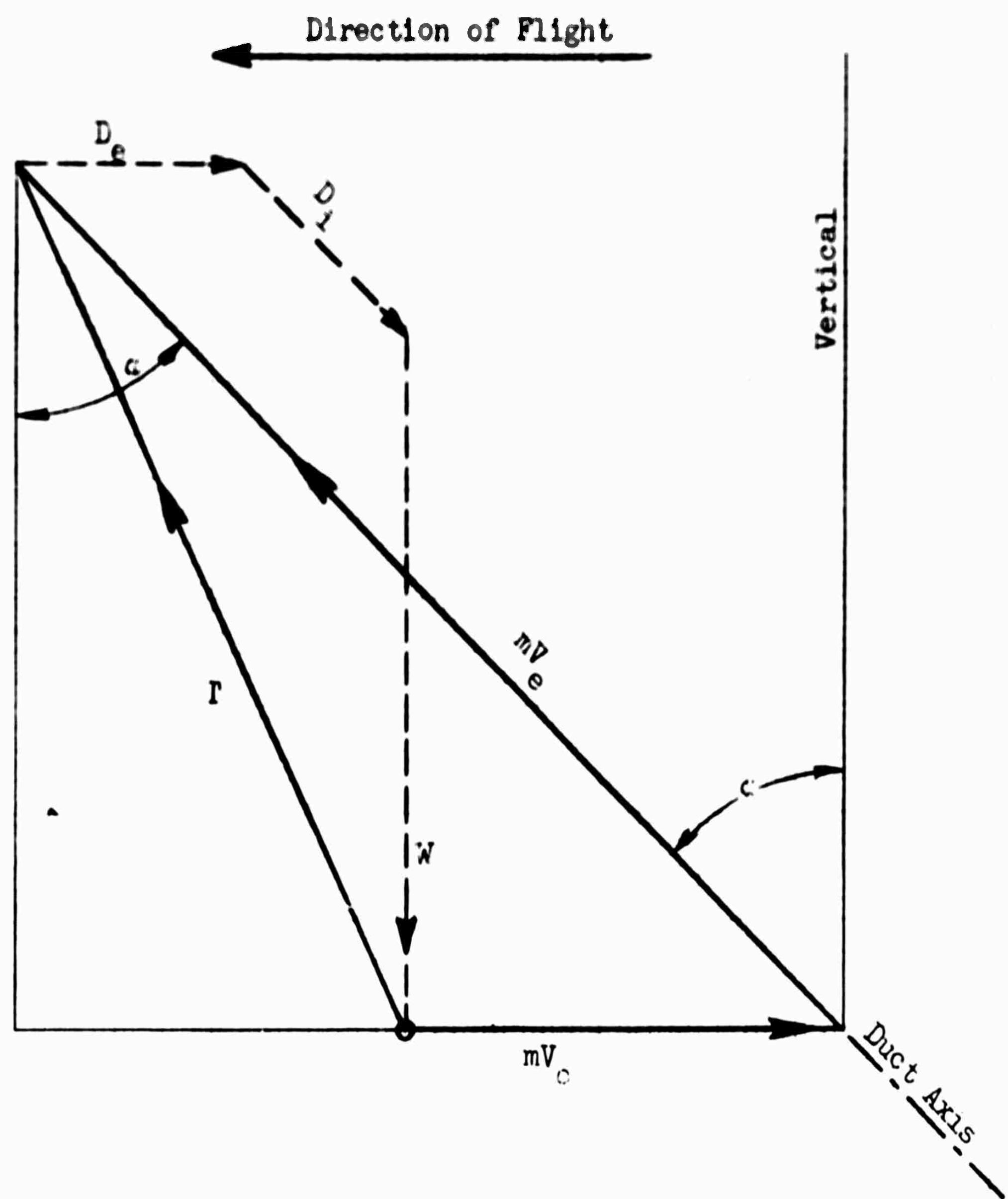


FIGURE 4: VECTOR DIAGRAM OF FORCES
(EQUILIBRIUM CONDITION, LEVEL FLIGHT)

FIGURE 5

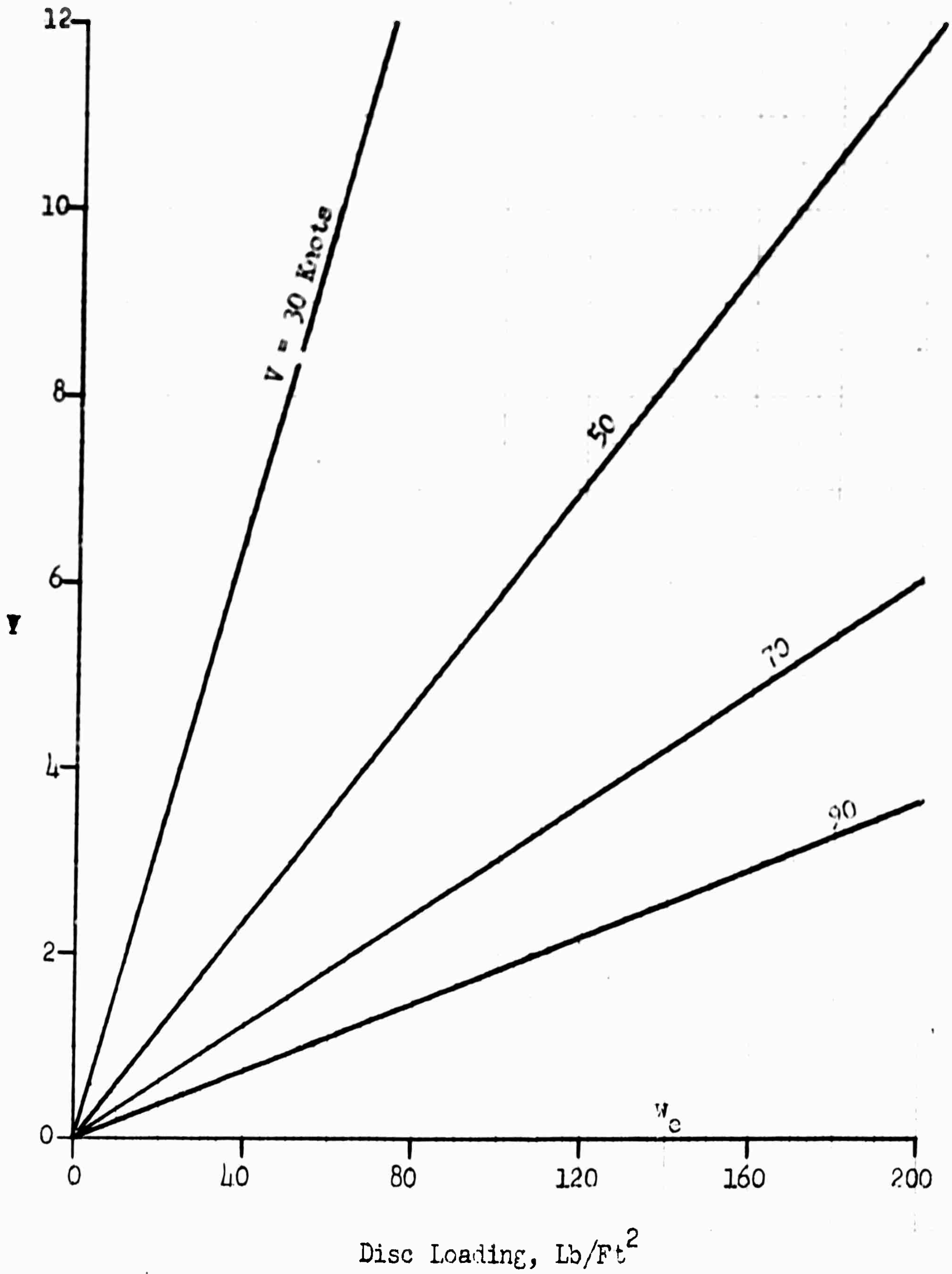


FIGURE 5: Ψ AS A FUNCTION OF DISC LOADING AND SPEED (SEA LEVEL)

$f_i = 0$

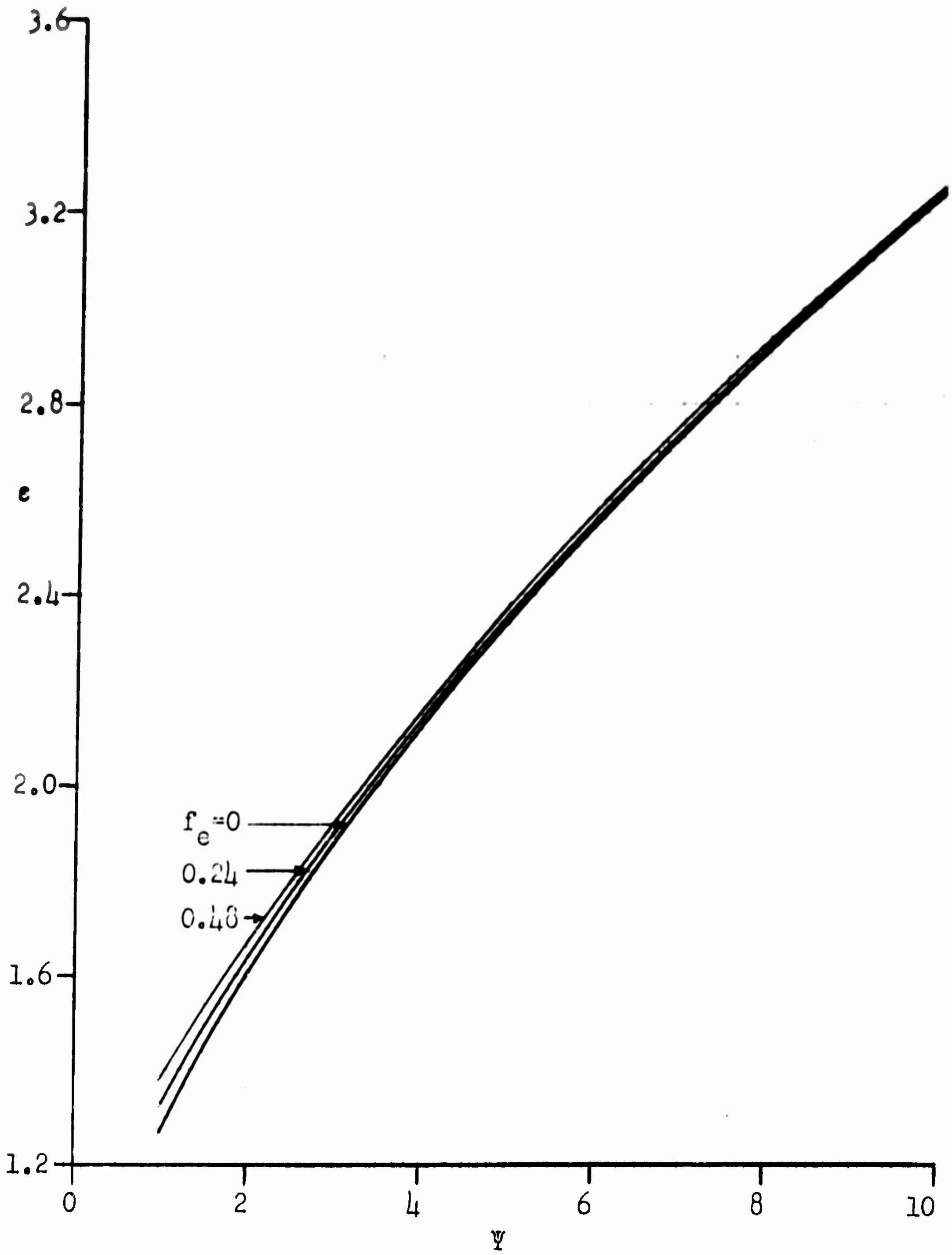


FIGURE 6: VELOCITY RATIO, ε vs Ψ (f_i = 0)

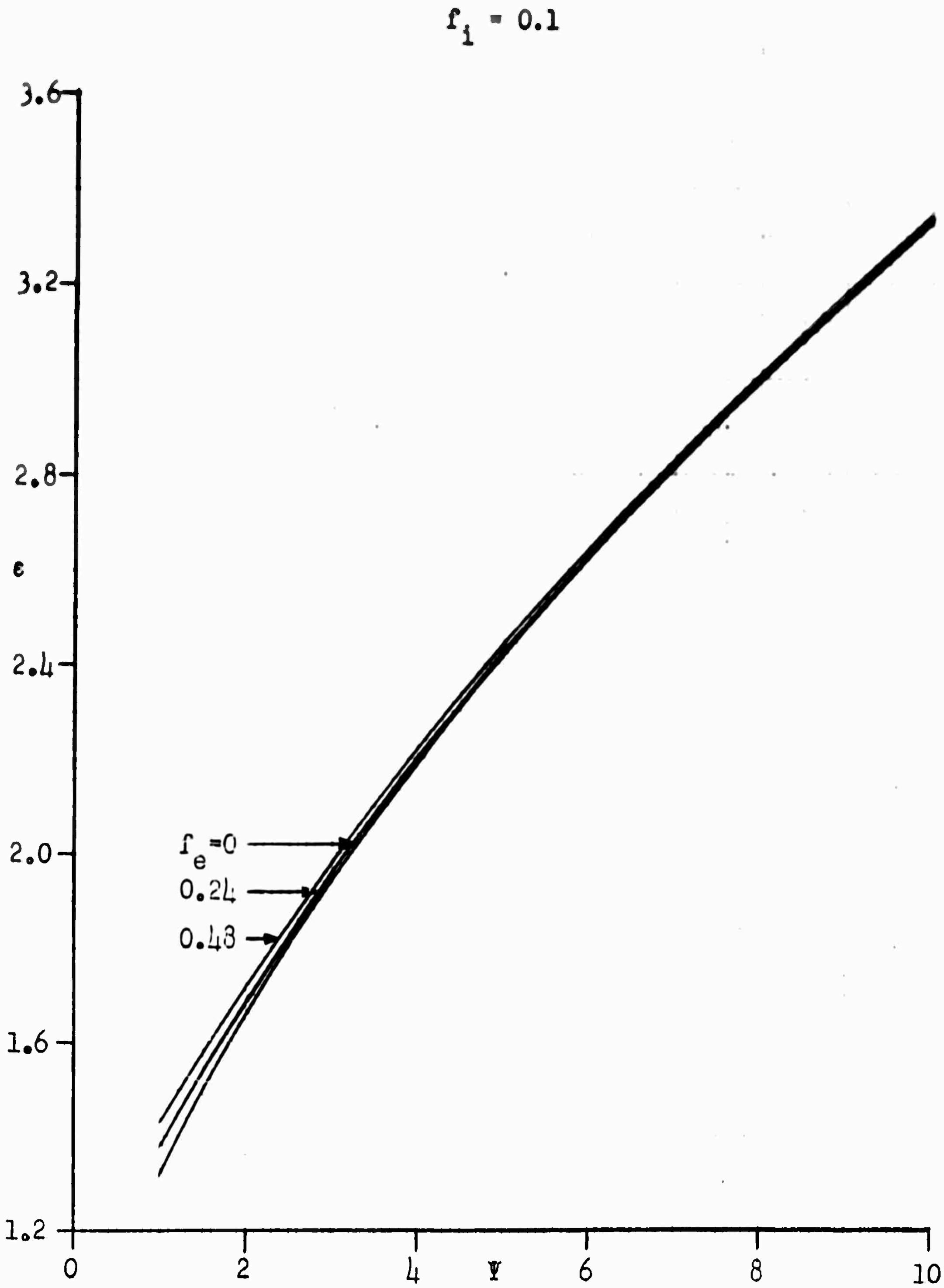


FIGURE 7: VELOCITY RATIO, ε vs Ψ (f_i = 0.1)

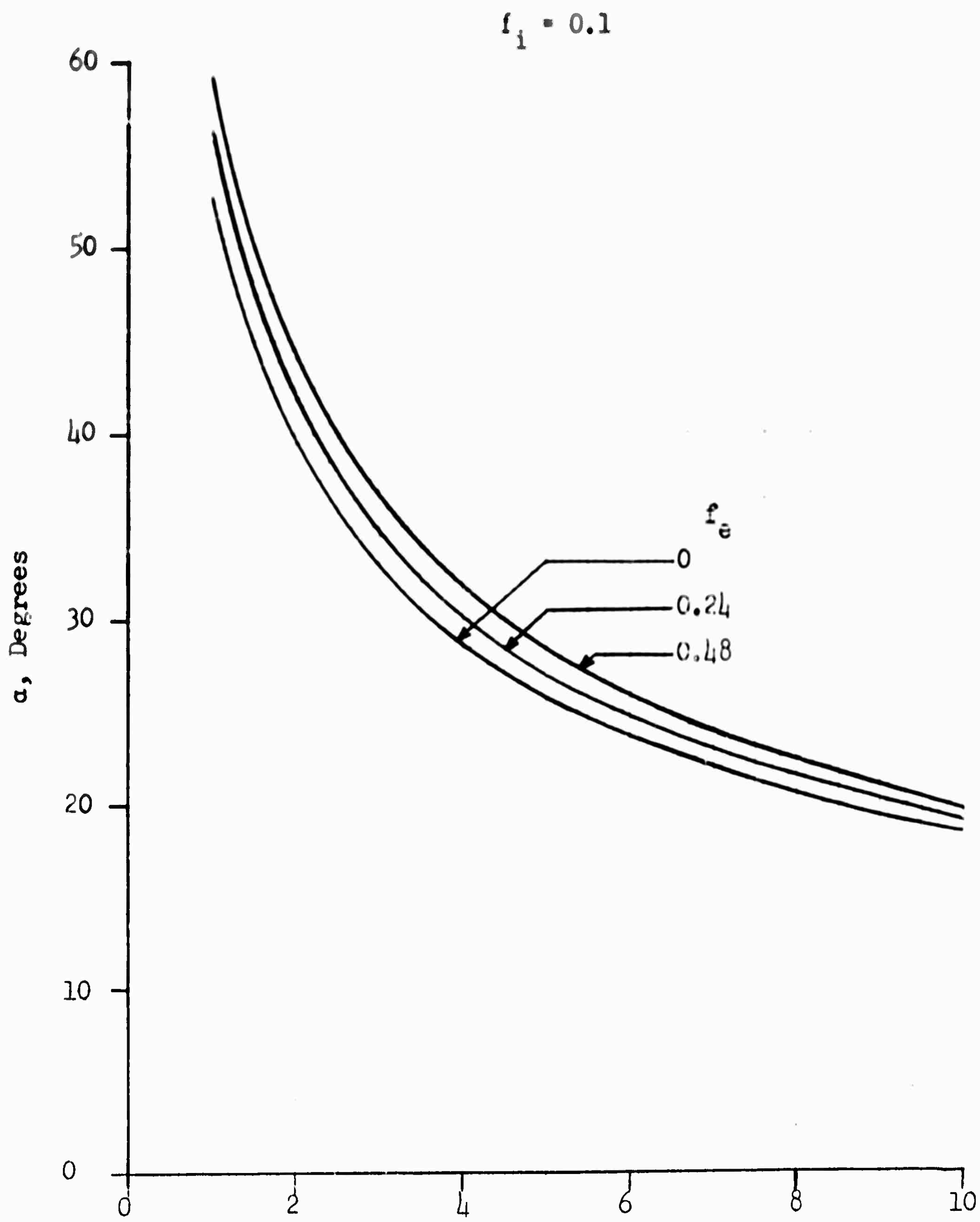


FIGURE 8: DUCT TILT ANGLE, α vs Ψ

FIGURE 9

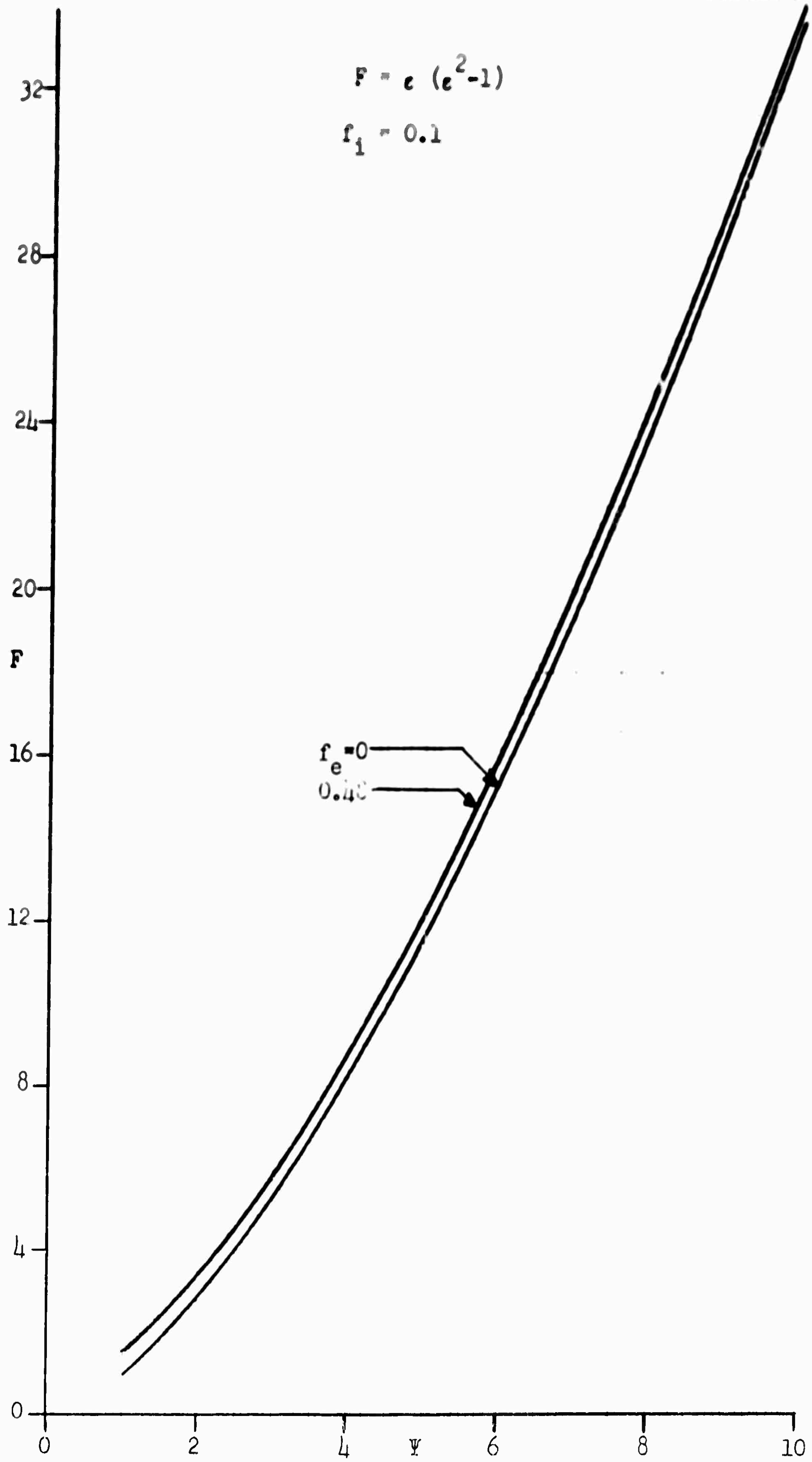


FIGURE 9: F vs Ψ

$$F' = \frac{dF}{d\Psi}$$

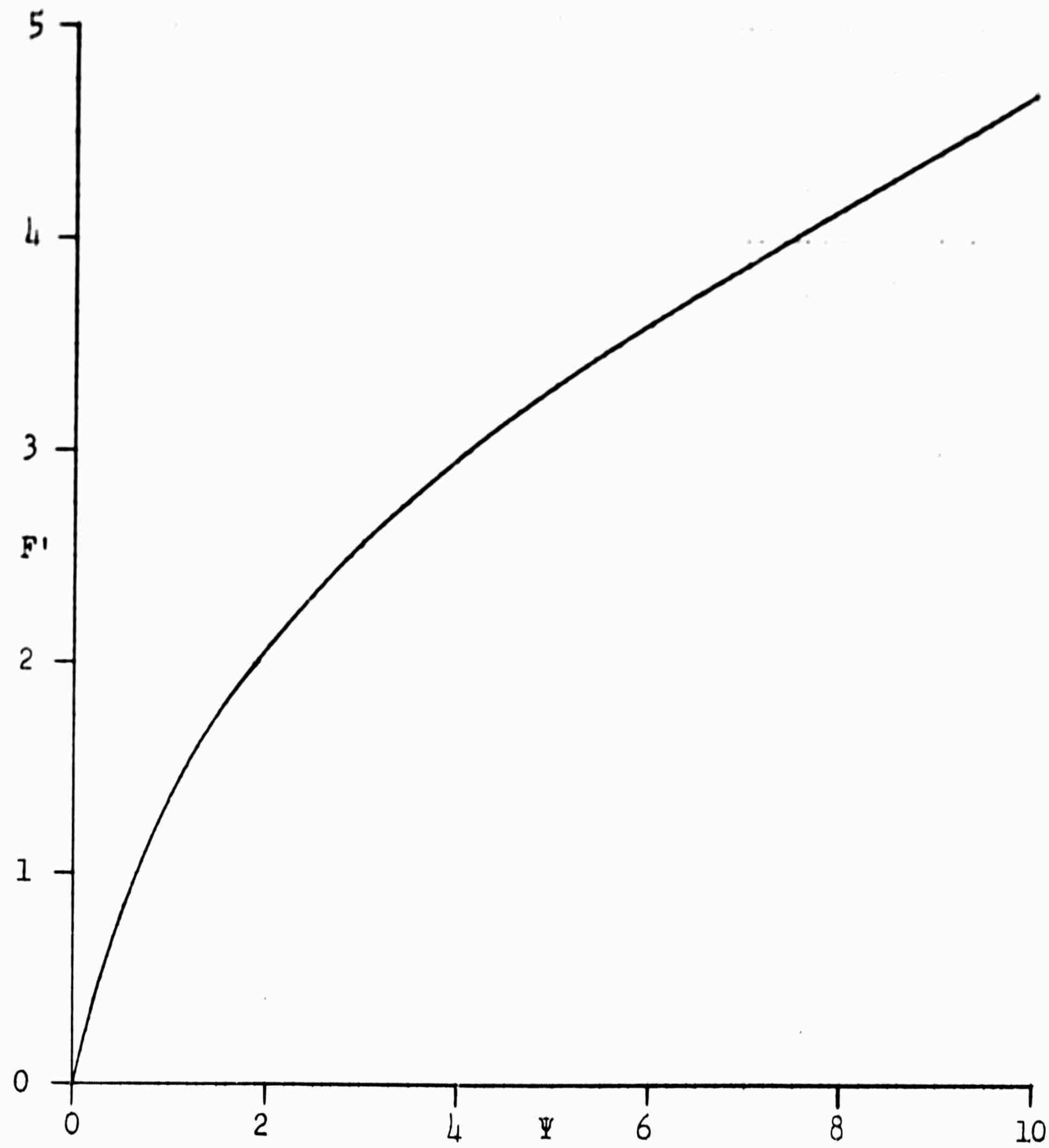


FIGURE 10: F' vs Ψ

FIGURE 11

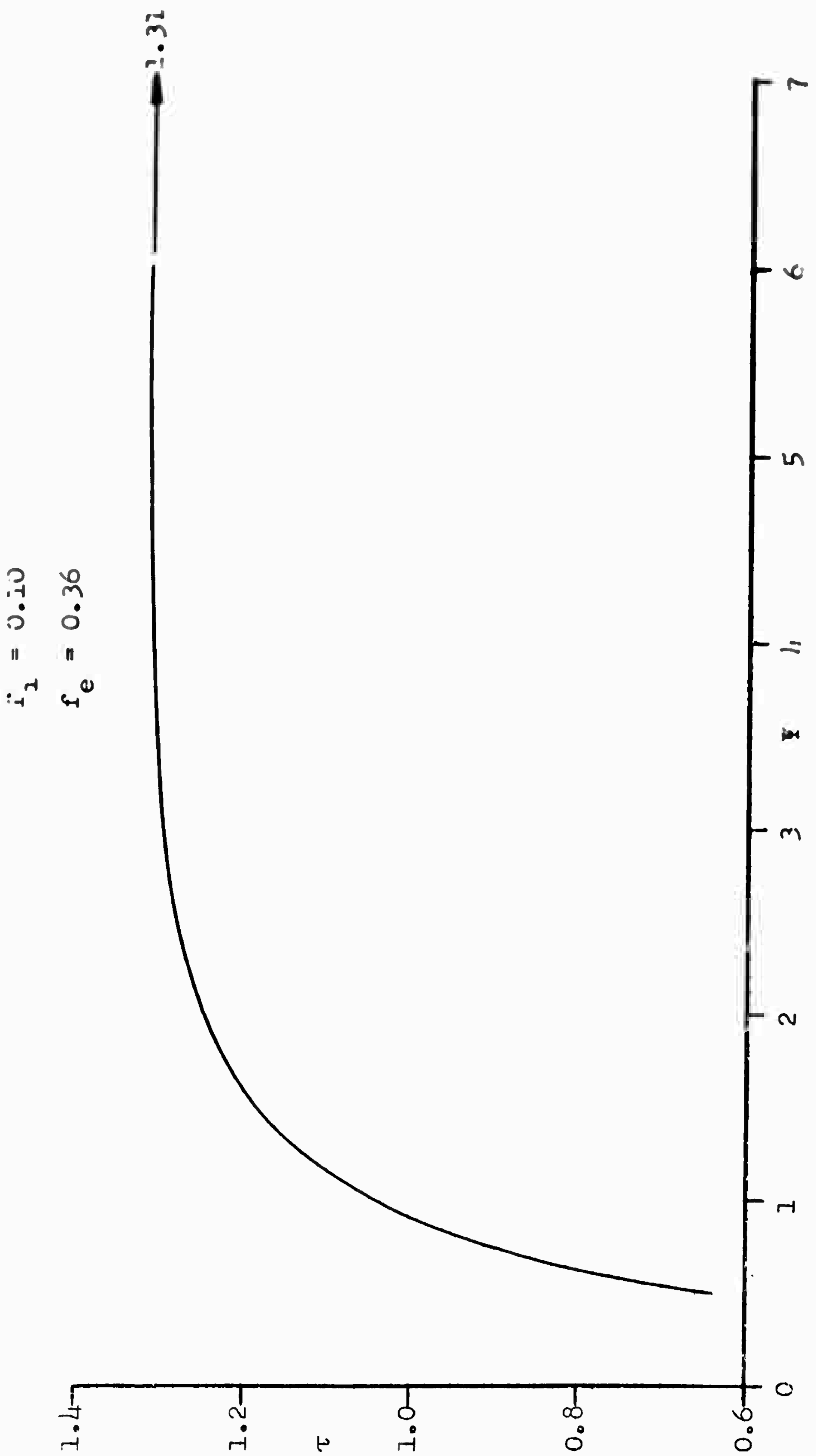


FIGURE 11: τ vs Ψ

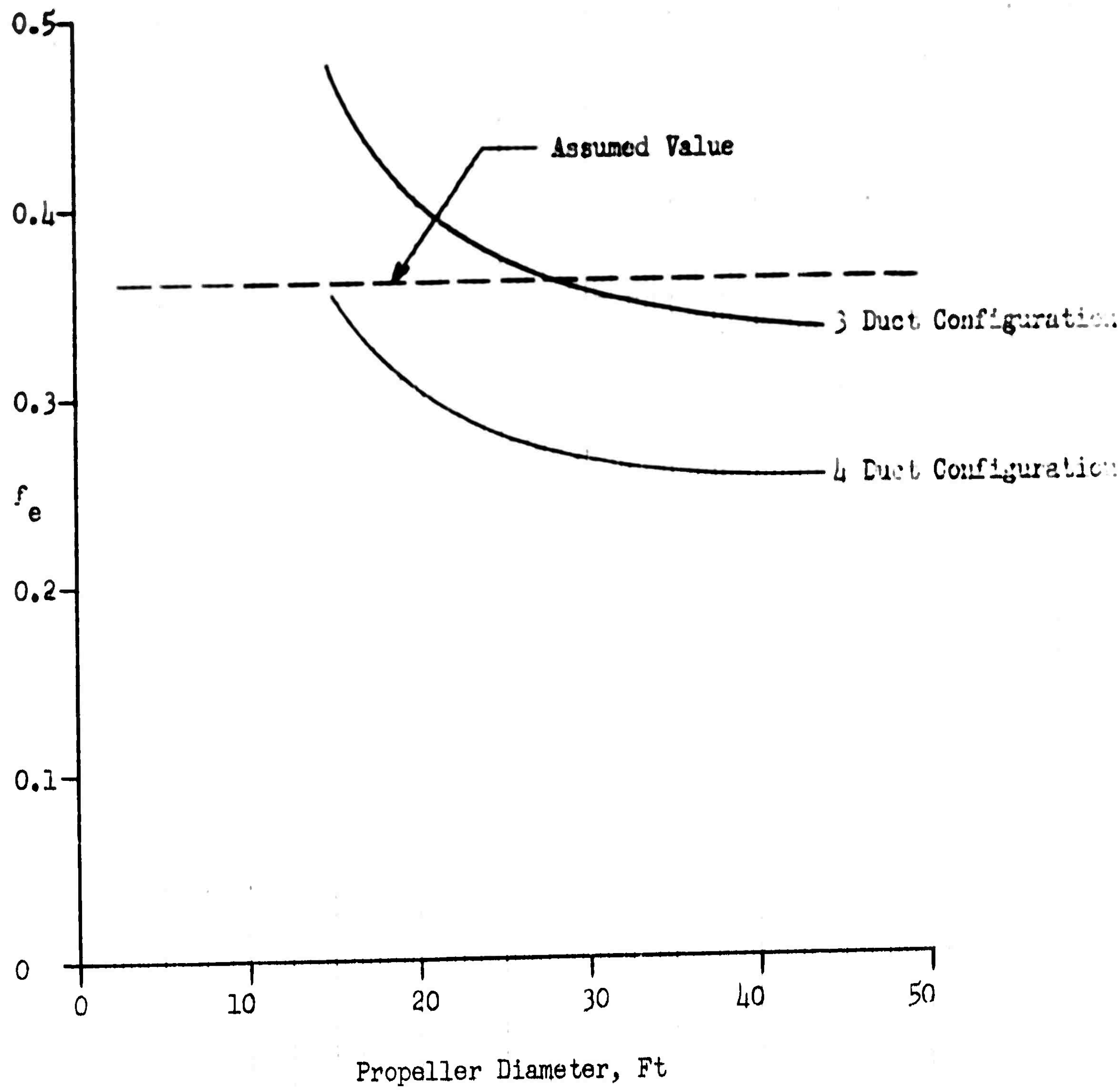


FIGURE 12: EXTERNAL DRAG COEFFICIENT

FIGURE 13

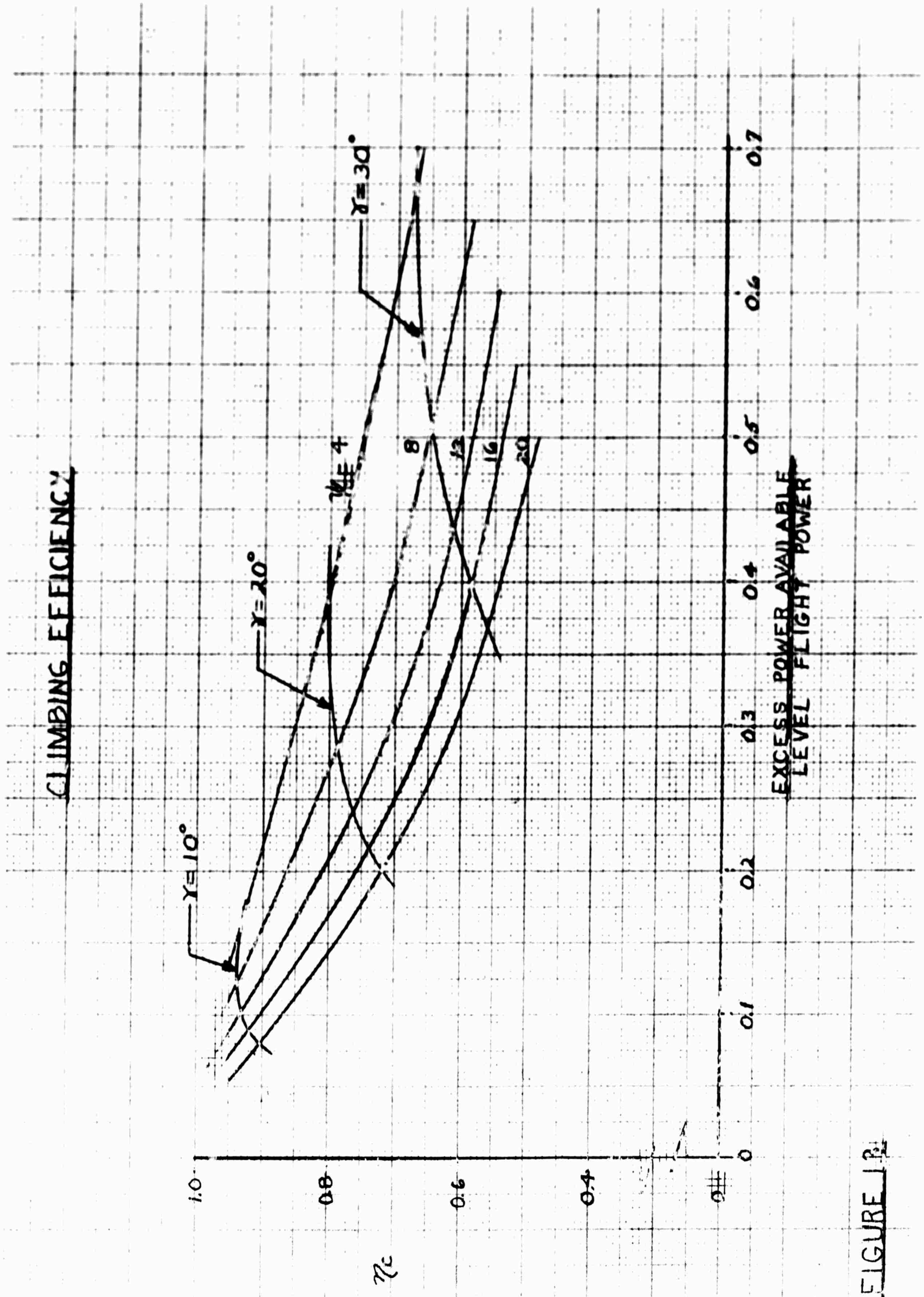


FIGURE 13

GAS TURBINE ENGINE CHARACTERISTICS

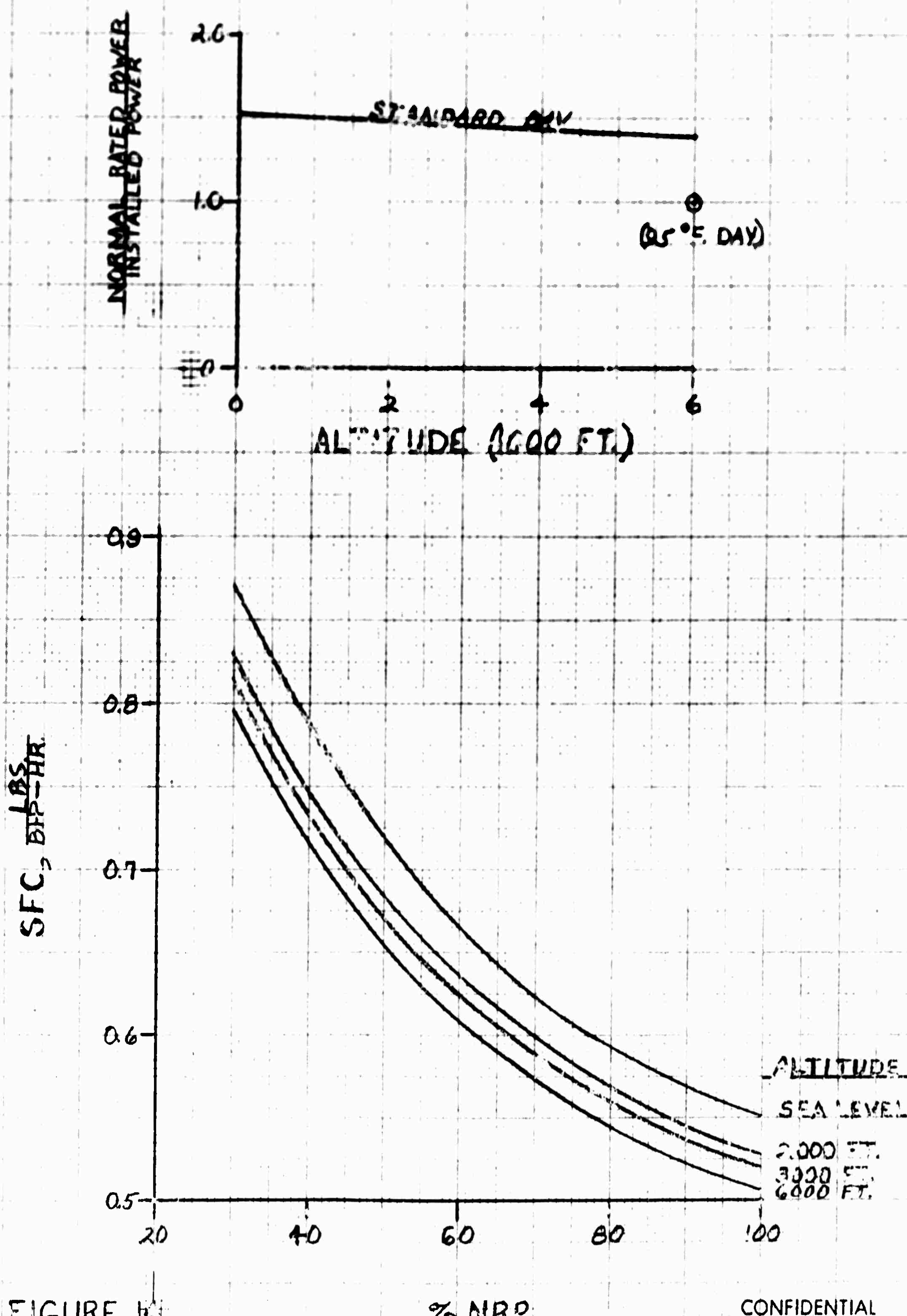


FIGURE 14

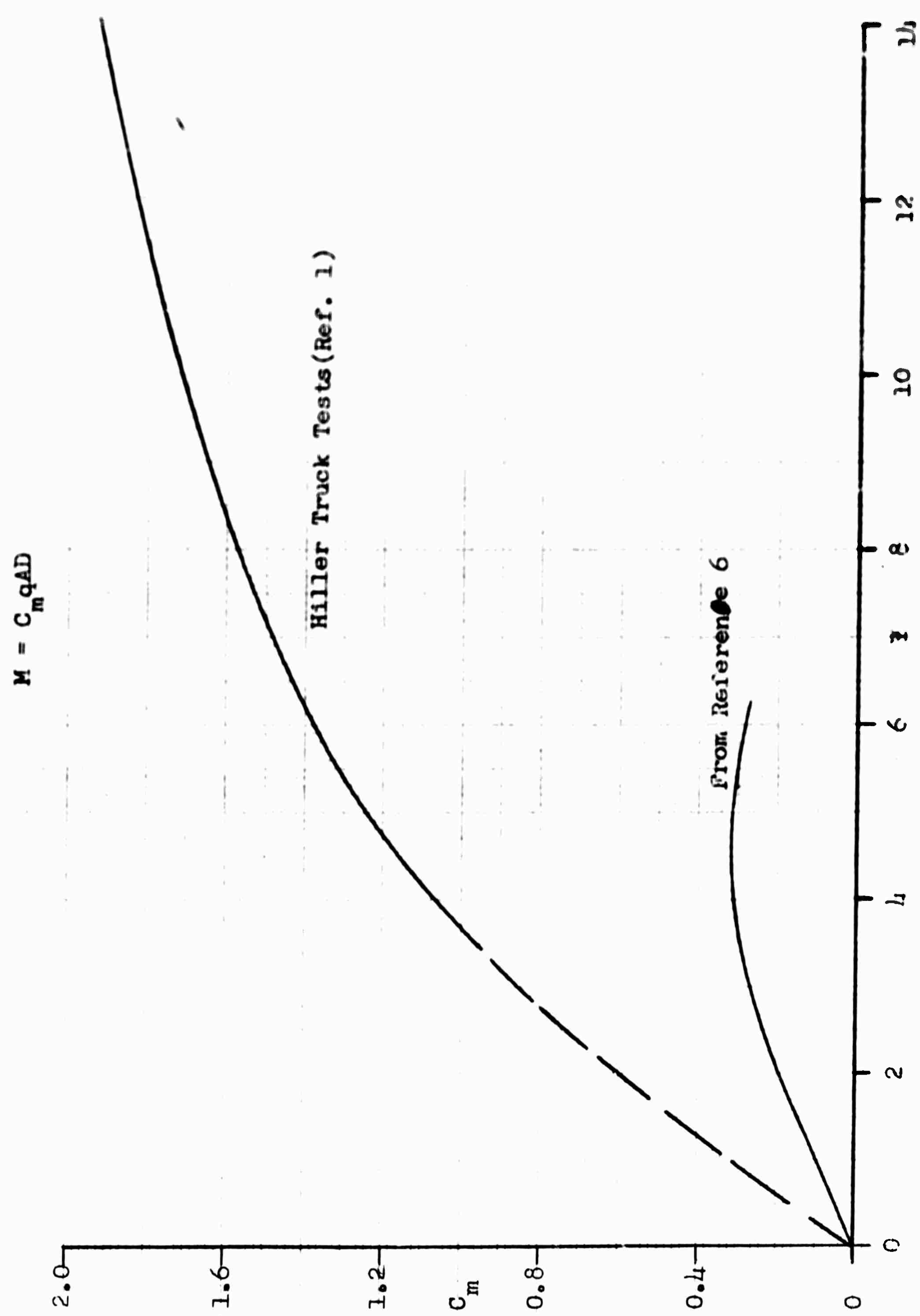


FIGURE 15: PITCHING MOMENT COEFFICIENT C_m vs Ψ

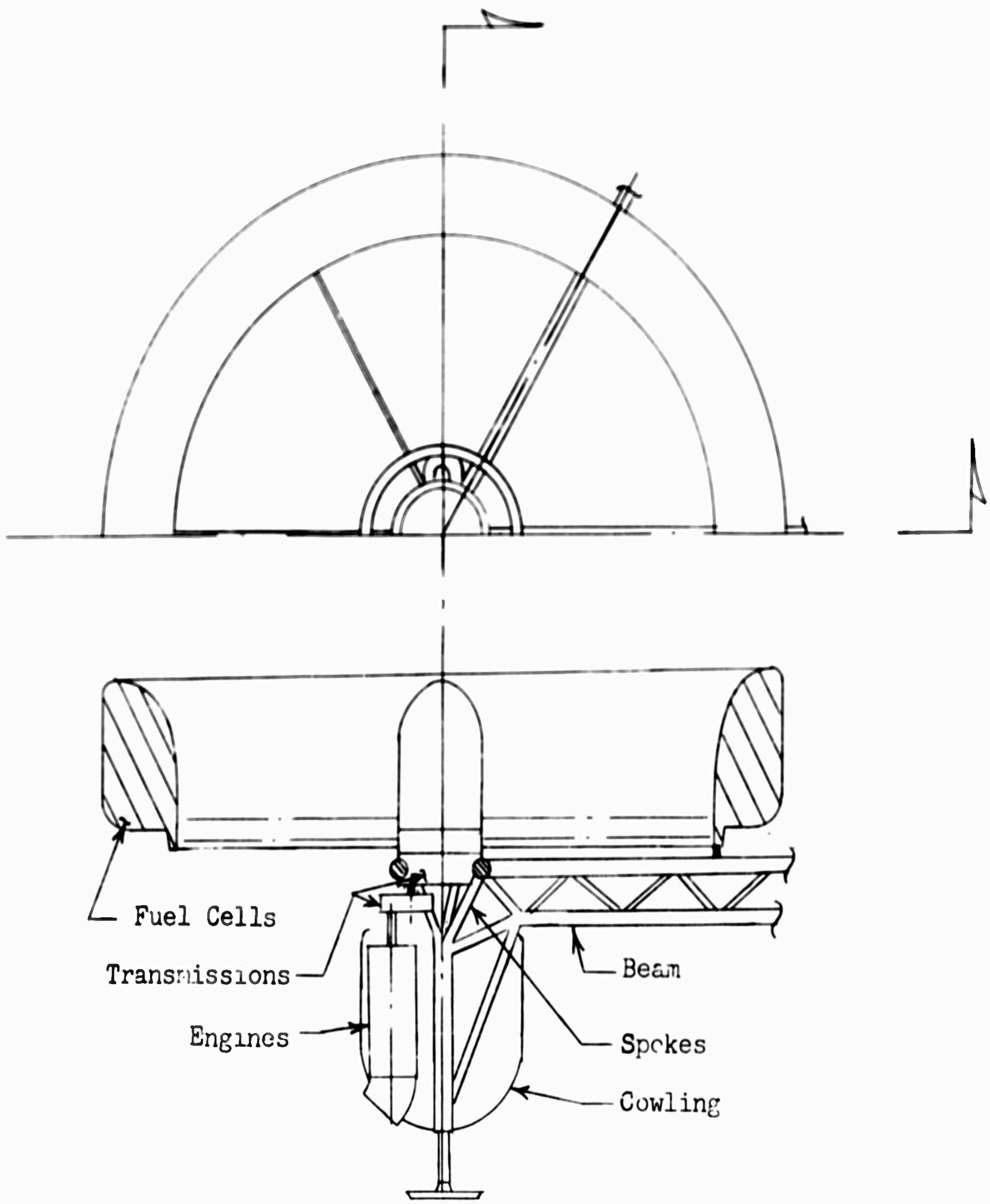


FIGURE 16: 3-DUCT NACELLE ARRANGEMENT

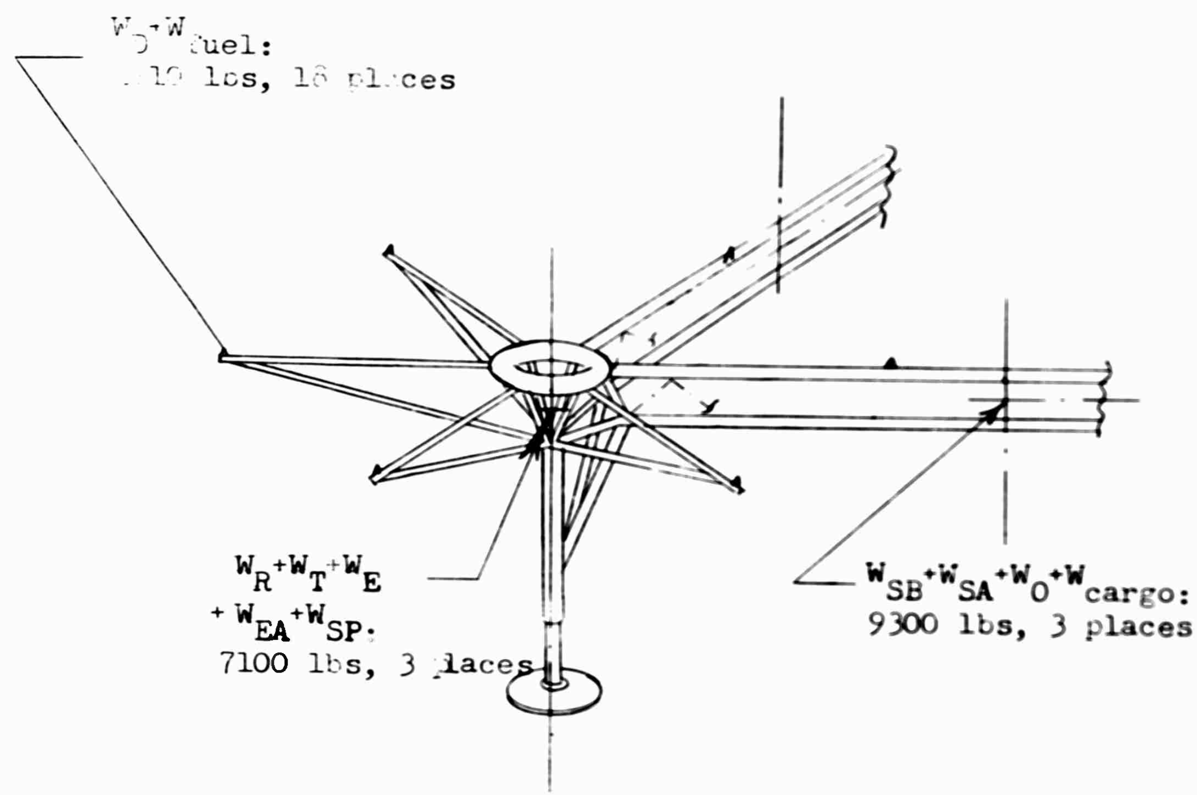
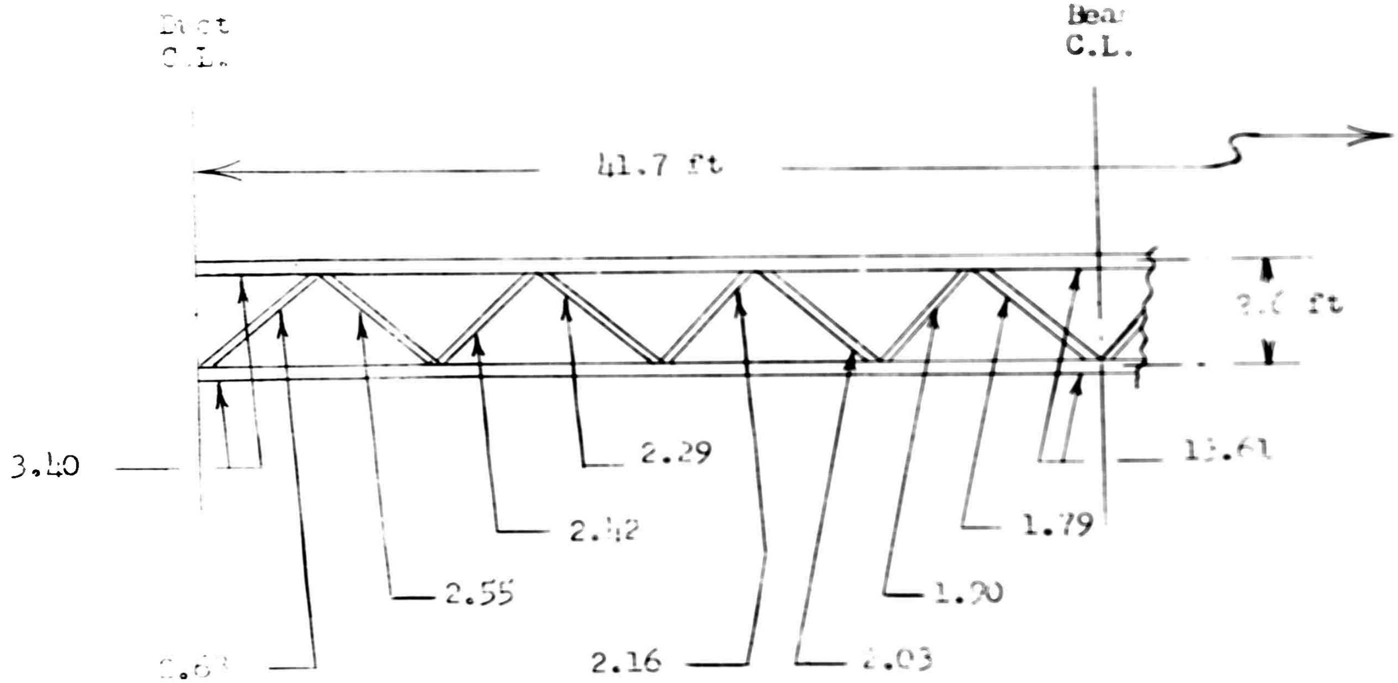


FIGURE 17: Disposition of Loads on 3-Duct Structure

W_G = [illegible],000 lbs, w = 35. psf



Cro[illegible]s [illegible]ection areas of [illegible]embers are [illegible] in [illegible]quare inches.
Longerons are tapered in uniform steps.

FIGURE 1 Provisional Bear for 3-Duct Configuration
W_G = [illegible] lbs, w [illegible] psf

FIGURE 19

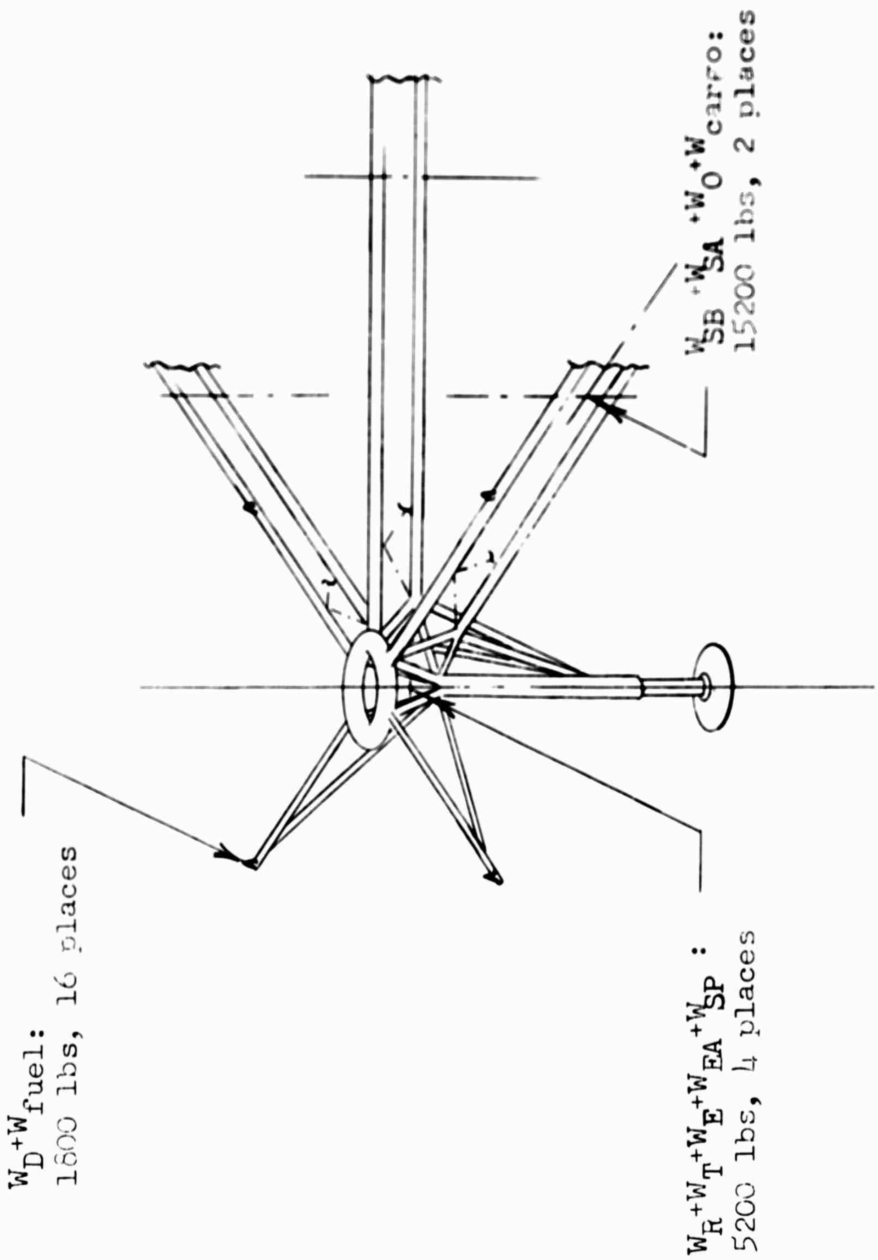
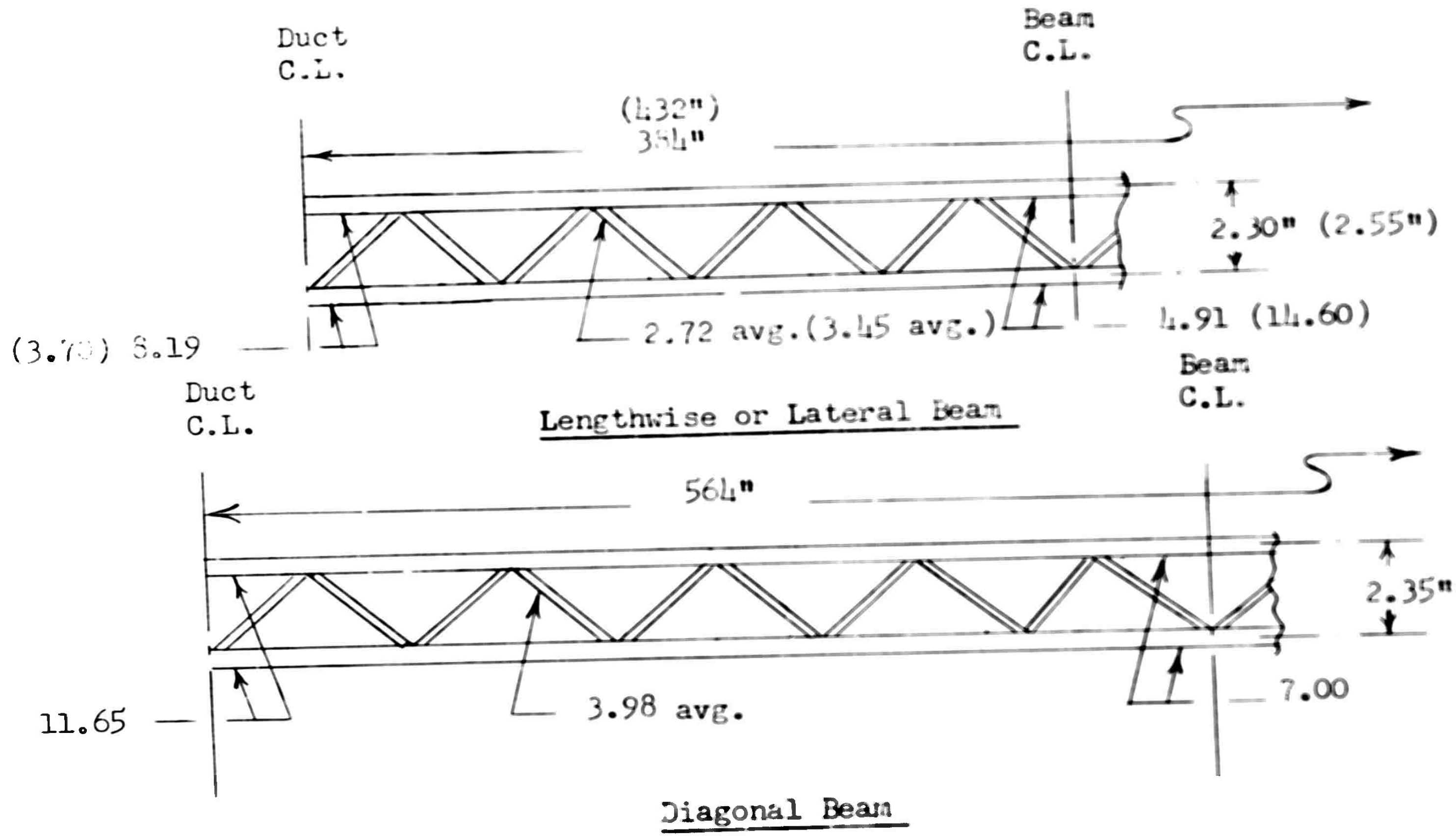


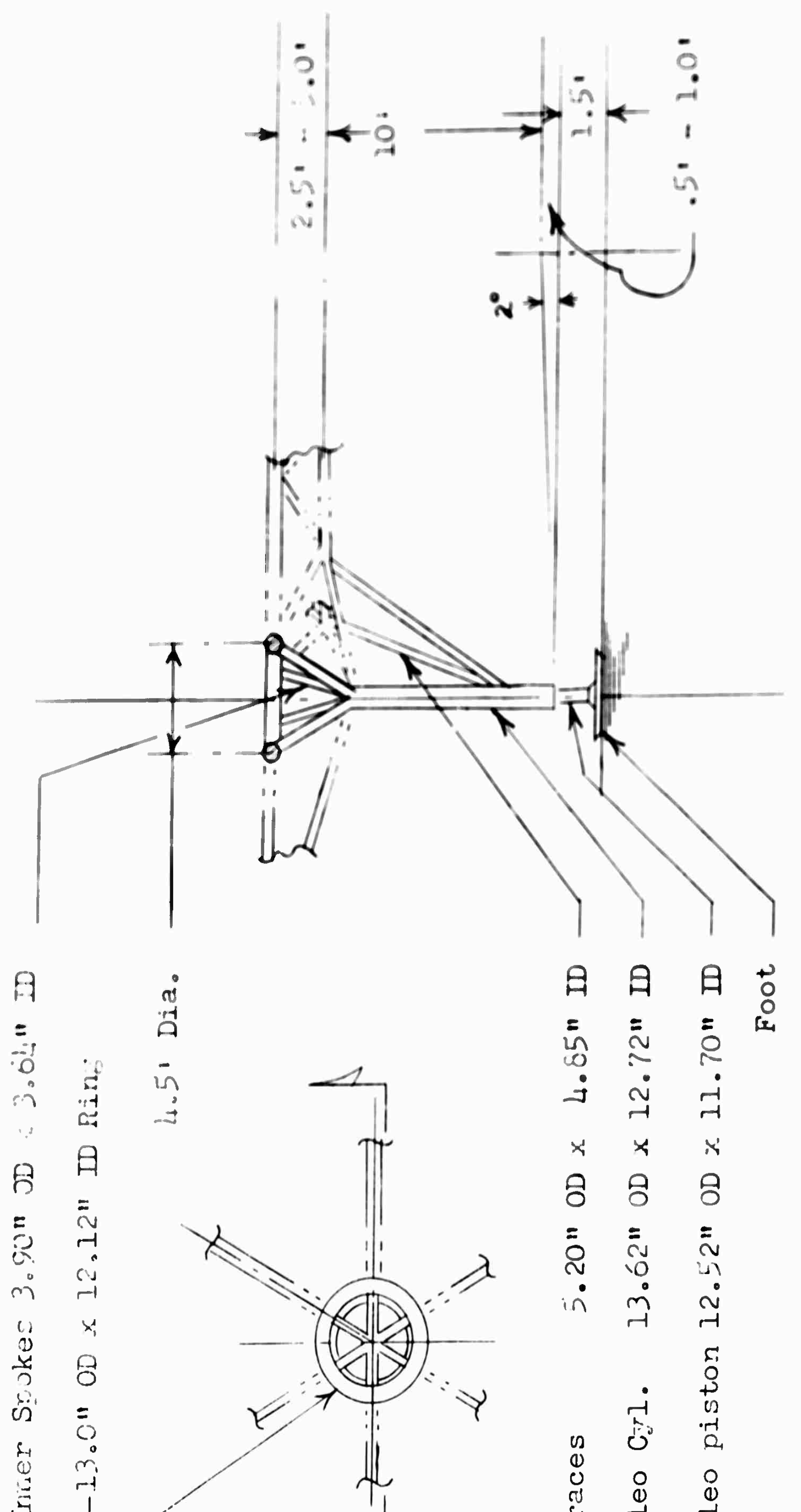
FIGURE 19: Disposition of Loads on 4-Duct Structure
W_G = 80,000 lbs, w = 35. psf



Member cross section areas shown in square inches. Dimensions in parenthesis are for Lateral Beams, which support the cargo winches.

FIGURE 20: Provisional Beam Design for 4-Duct Configuration
W_G = 80,000 lbs, w = 35. psf

FIGURE 21



Note: Outer spokes are included in Duct Weight

FIGURE 21: Provisional Design of Pylon for 3-Duct Configuration

W_G = [illegible]0,000 lbs, w = 35. psf

EFFECT OF RADIUS

HOVER TIME, t_H = 15 MIN.
DISC LOADING, w = 75 LBS/FT.^{2}
PAYLOAD P = 24,000 LBS.

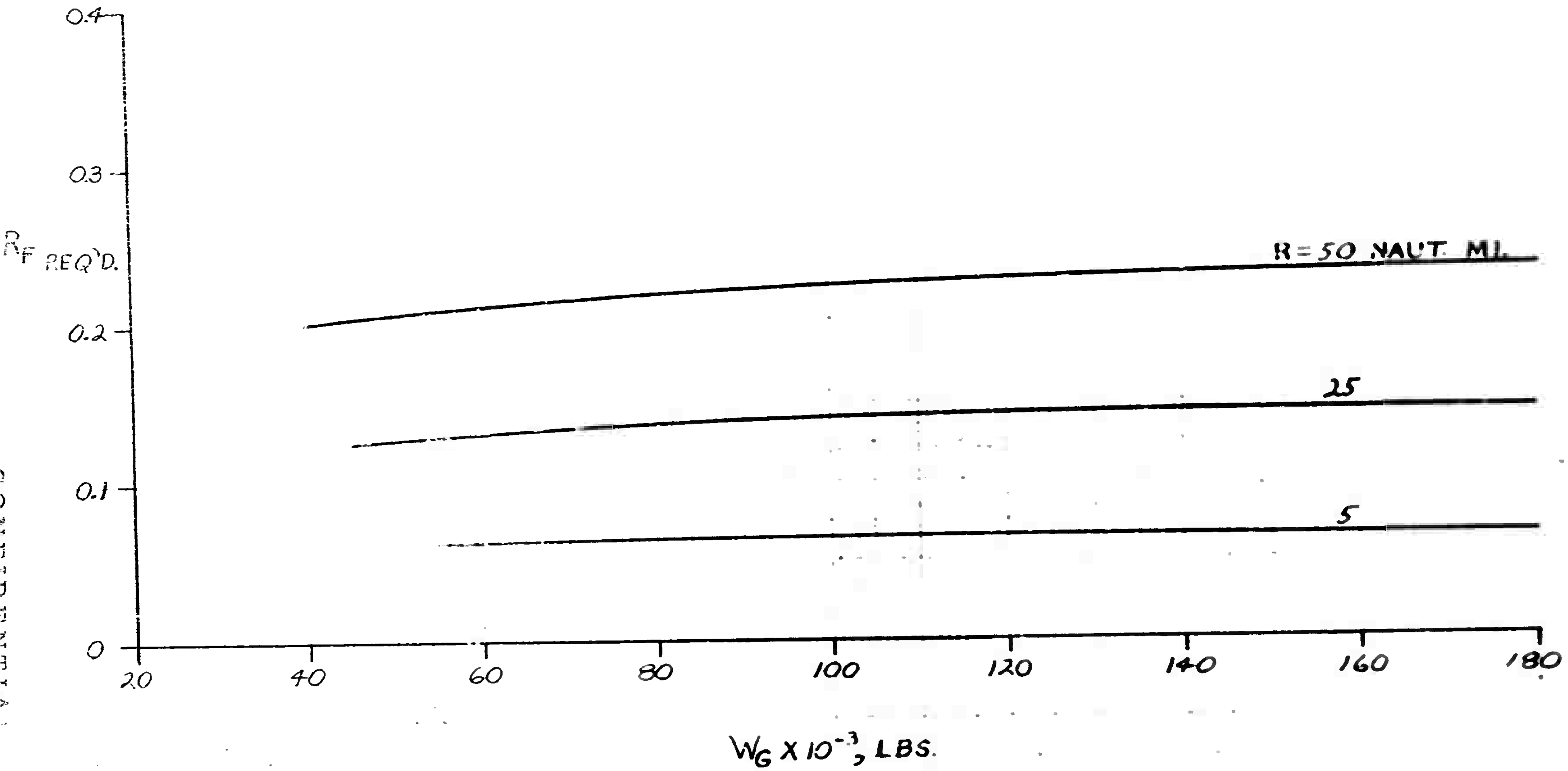


FIGURE 23

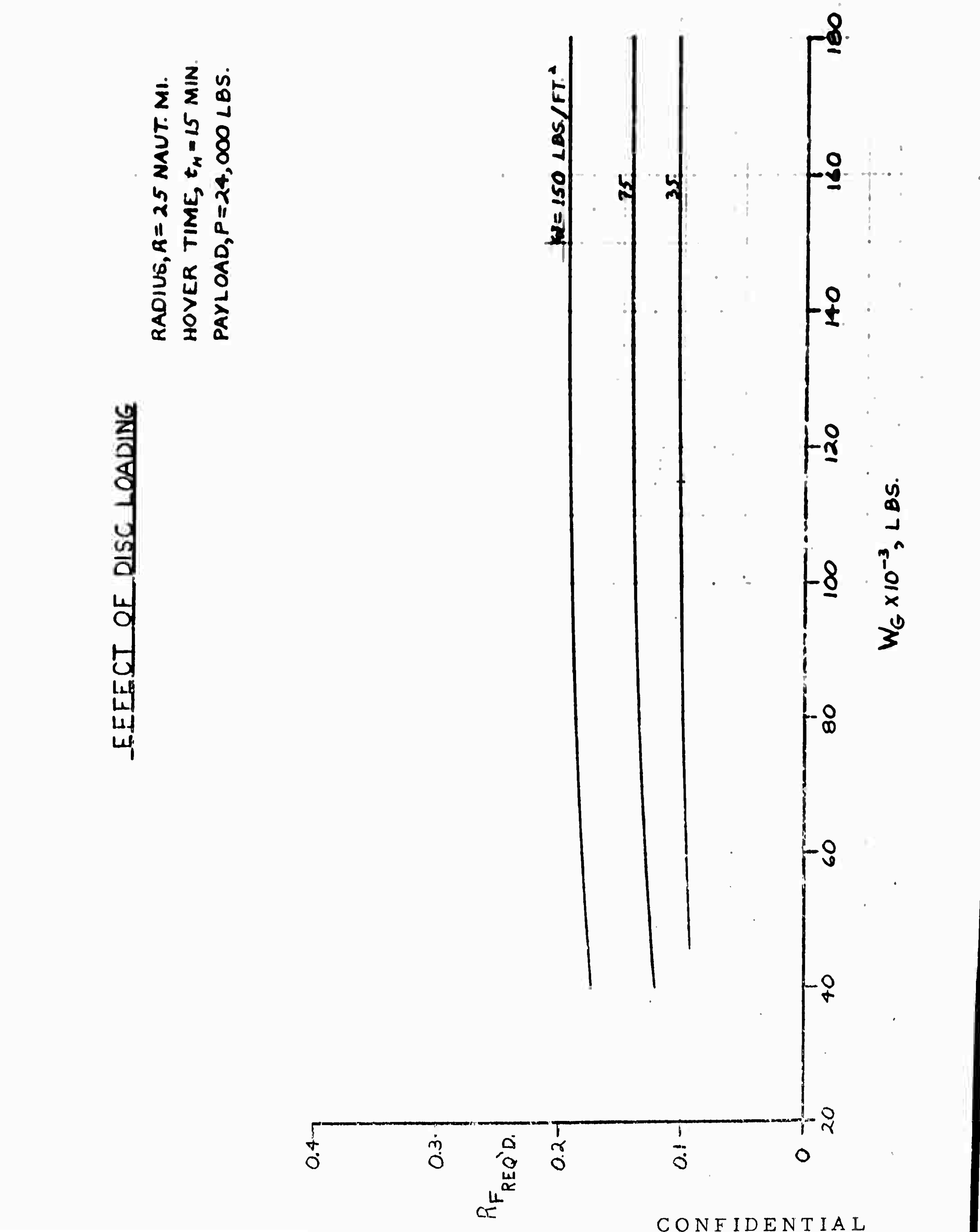
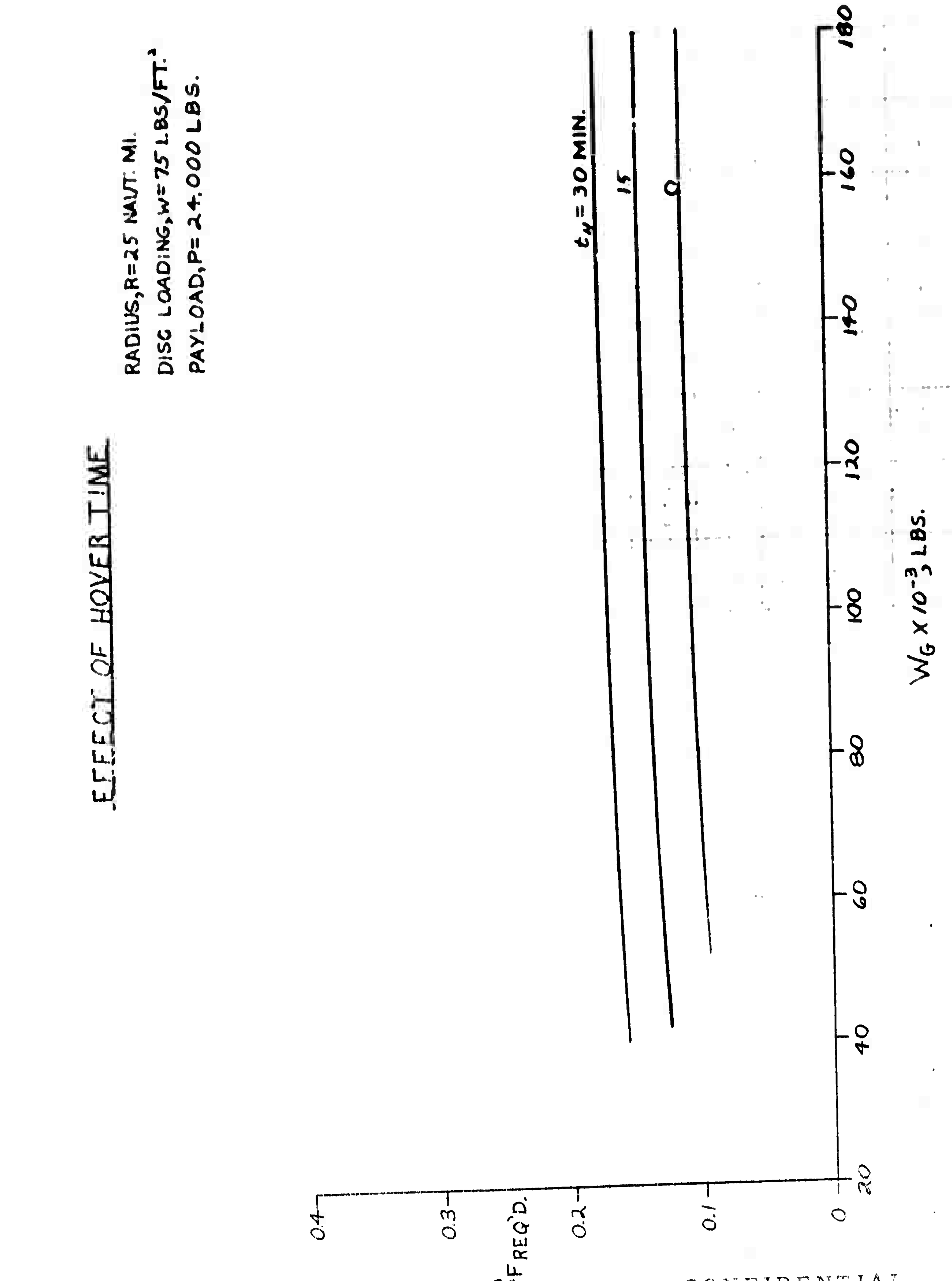
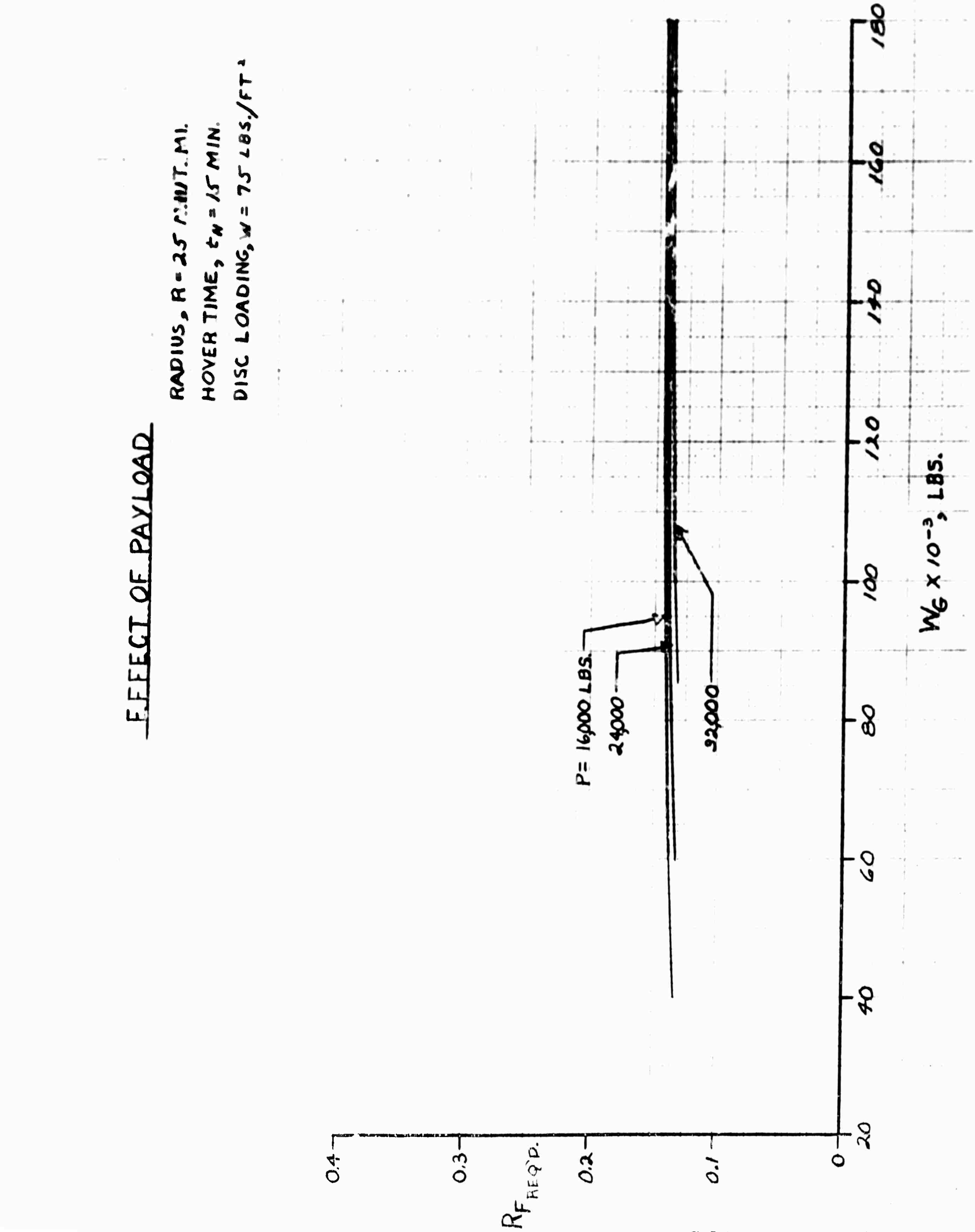

EFFECT OF DISC LOADING
RADIUS, R = 25 NAUT. MI.
HOVER TIME, t_H = 15 MIN.
PAYLOAD, P = 24,000 LBS.
W/A = 150 LBS/FT.^{2}
75
35
0.4
0.3
0.2
0.1
0
$R_{F_{REQ'D.}}$
20
40
60
80
100
120
140
160
180
$W_G \times 10^{-3}$, LBS.

FIGURE 24


EFFECT OF HOVER TIME
RADIUS, R=25 NAUT. MI.
DISC LOADING, w=75 LBS/FT.²
PAYLOAD, P=24,000 LBS.
t_H = 30 MIN.
15
0
$R_{F\,REQ'D.}$
0
0.1
0.2
0.3
0.4
20
40
60
80
100
120
140
160
180
$W_G \times 10^{-3}$, LBS.

ARO [illegible] CONTRACT DA [illegible]

FIGURE 25



EFFECT OF DISC LOADING

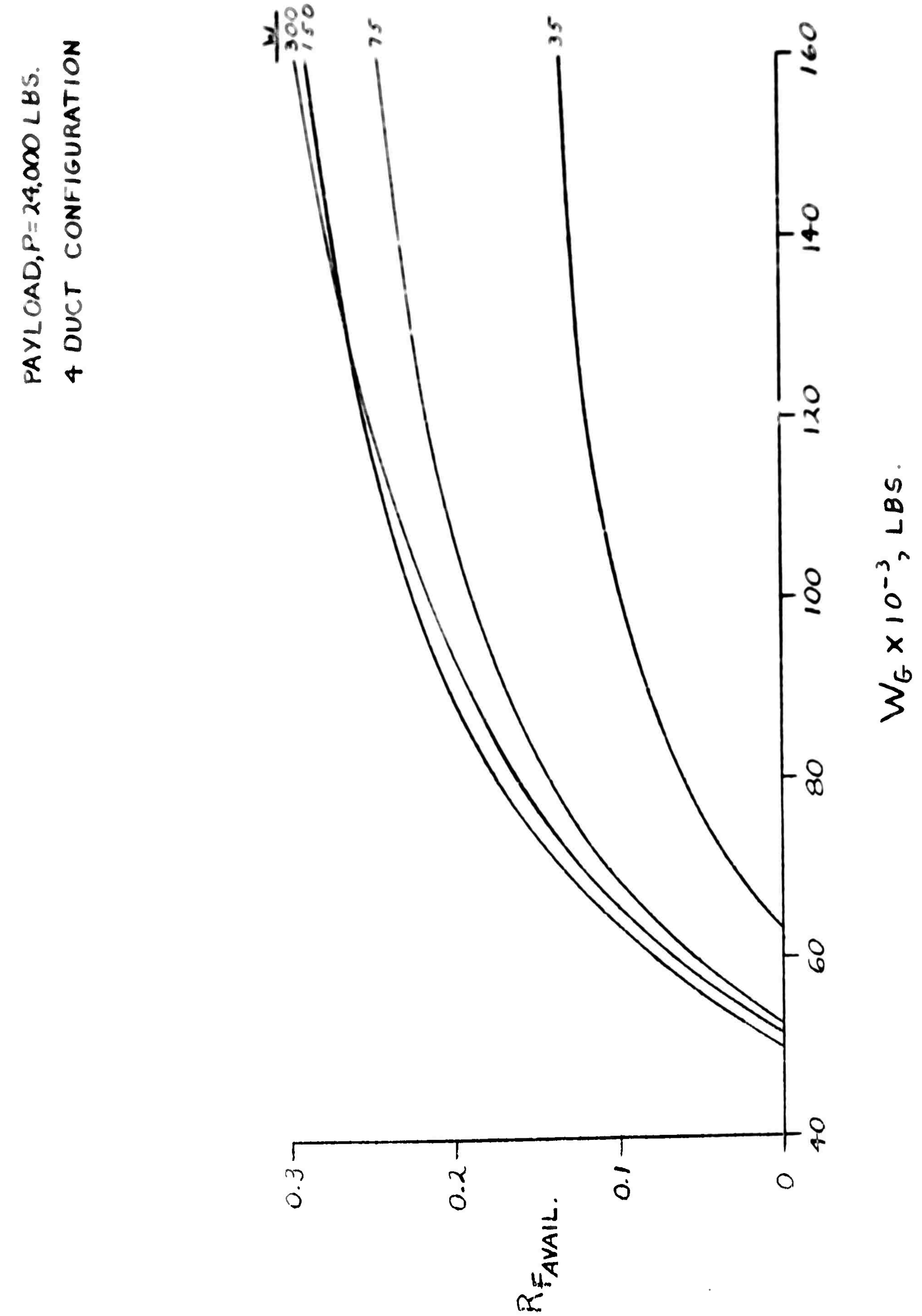
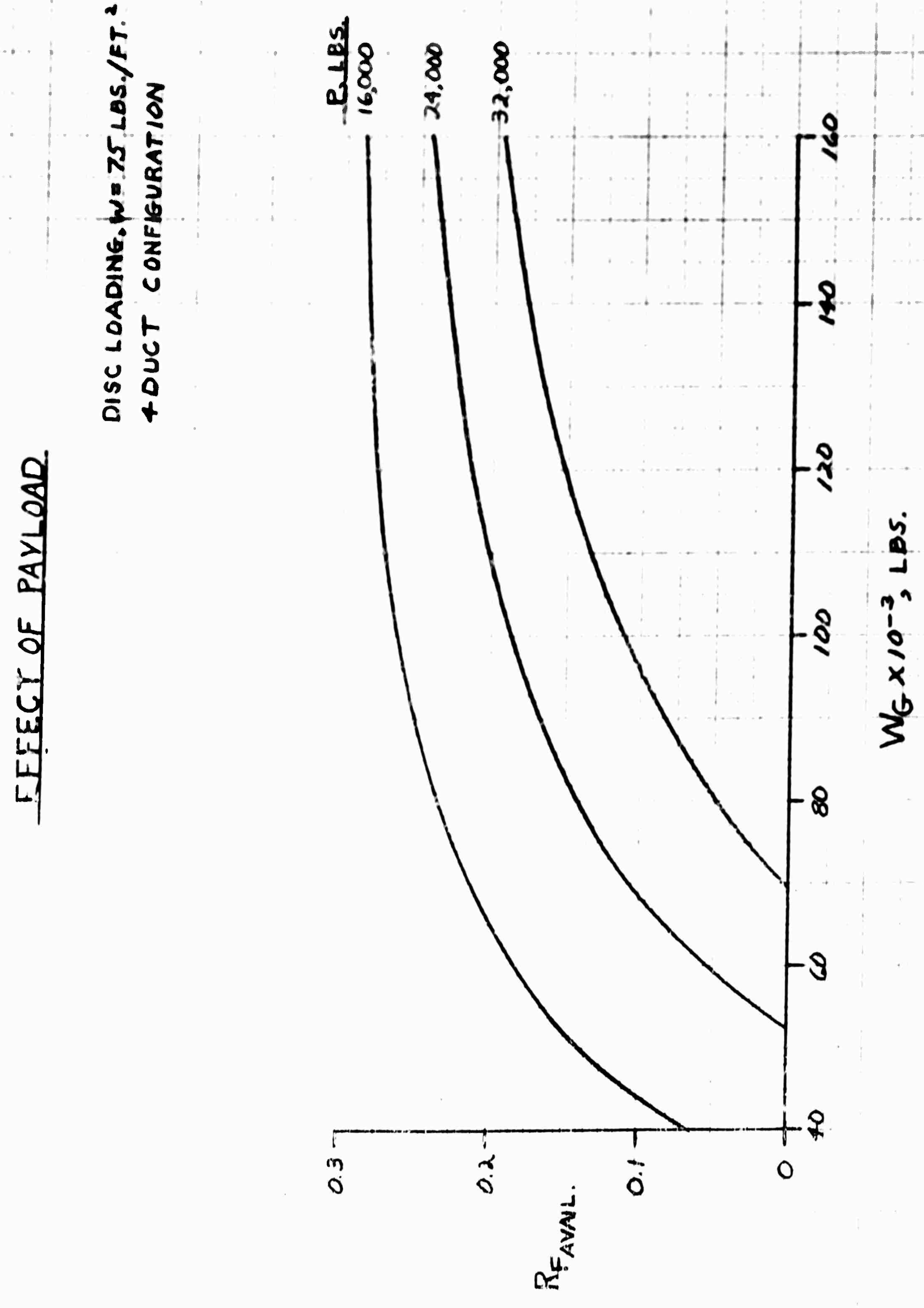


FIGURE 27


EFFECT OF PAYLOAD
DISC LOADING, w = 75 LBS./FT.²
4 DUCT CONFIGURATION
P, LBS.
16,000
24,000
32,000
$R_{F_{AVAIL.}}$
0.3
0.2
0.1
0
40
60
80
100
120
140
160
$W_G \times 10^{-3}$, LBS.

EFFECT OF DUCT CONFIGURATION

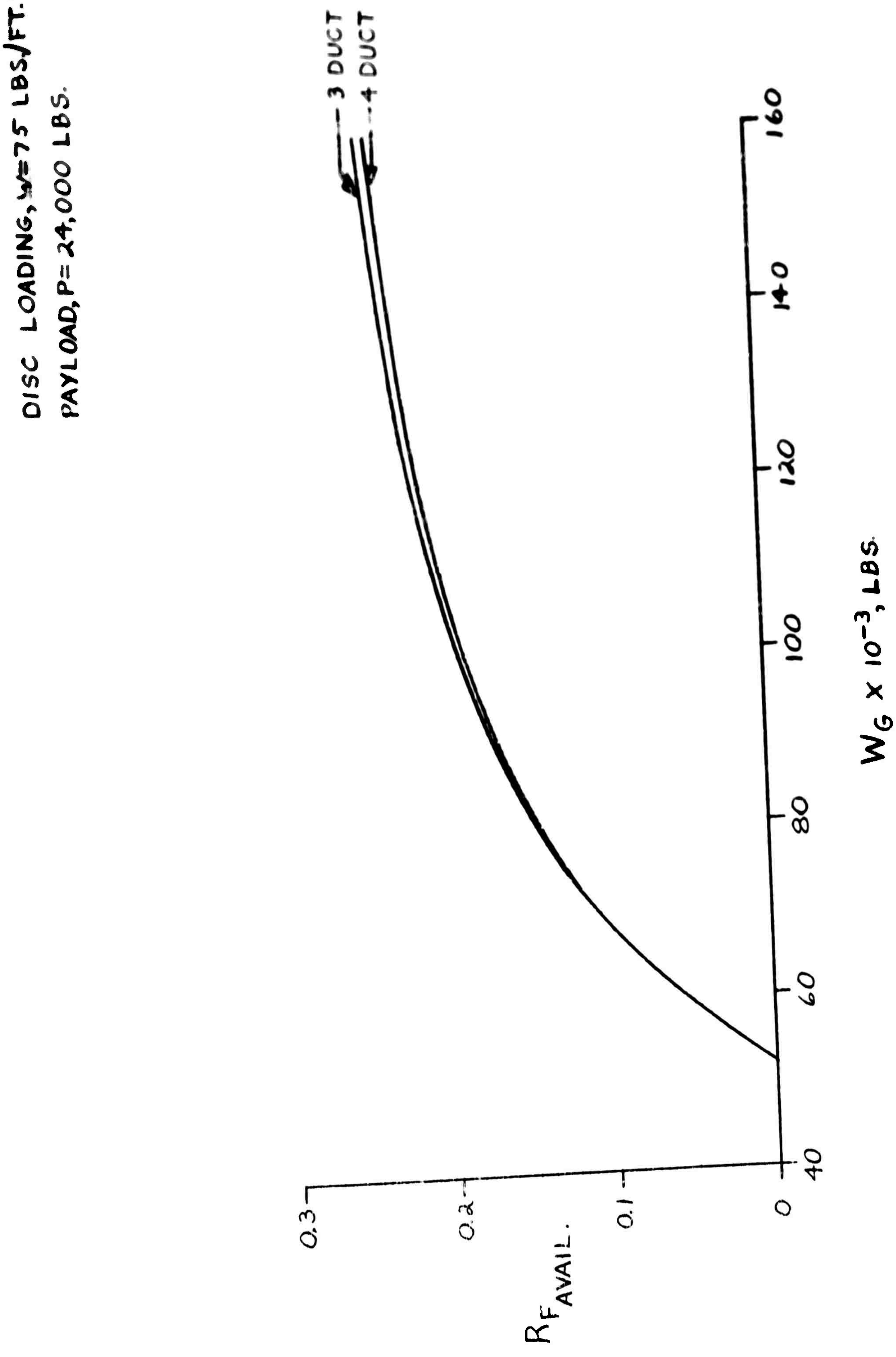
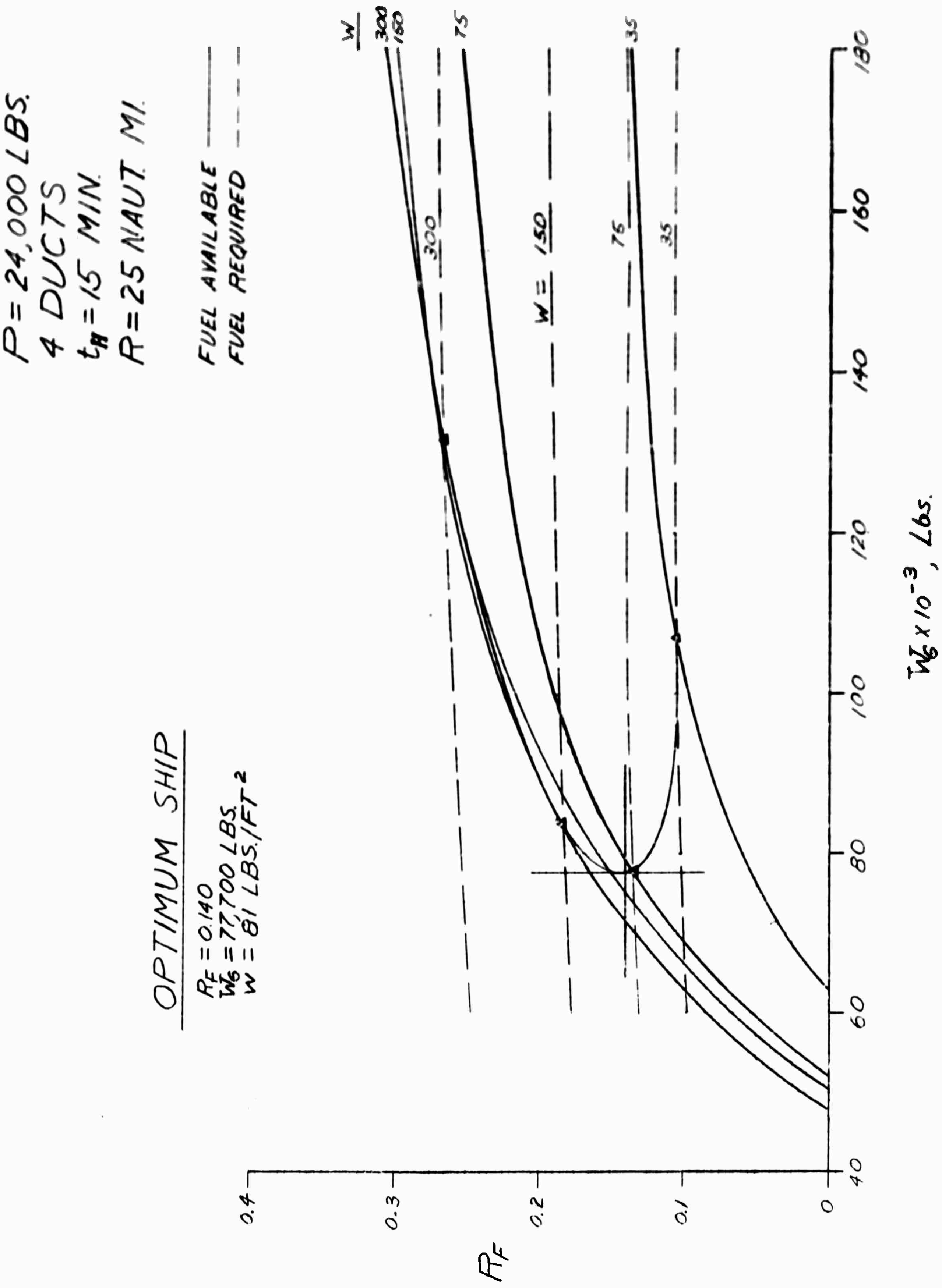


FIGURE 29


P = 24,000 LBS.
4 DUCTS
t_H = 15 MIN.
R = 25 NAUT. MI.
FUEL AVAILABLE
FUEL REQUIRED
OPTIMUM SHIP
R_F = 0.140
W_G = 77,700 LBS.
w = 81 LBS./FT²
w
300
180
75
35
W = 150
300
75
35
0.4
0.3
0.2
0.1
0
R_F
40
60
80
100
120
140
160
180
$W_G \times 10^{-3}$, Lbs.

OPTIMUM SHIPS
EFFECT OF PAYLOAD

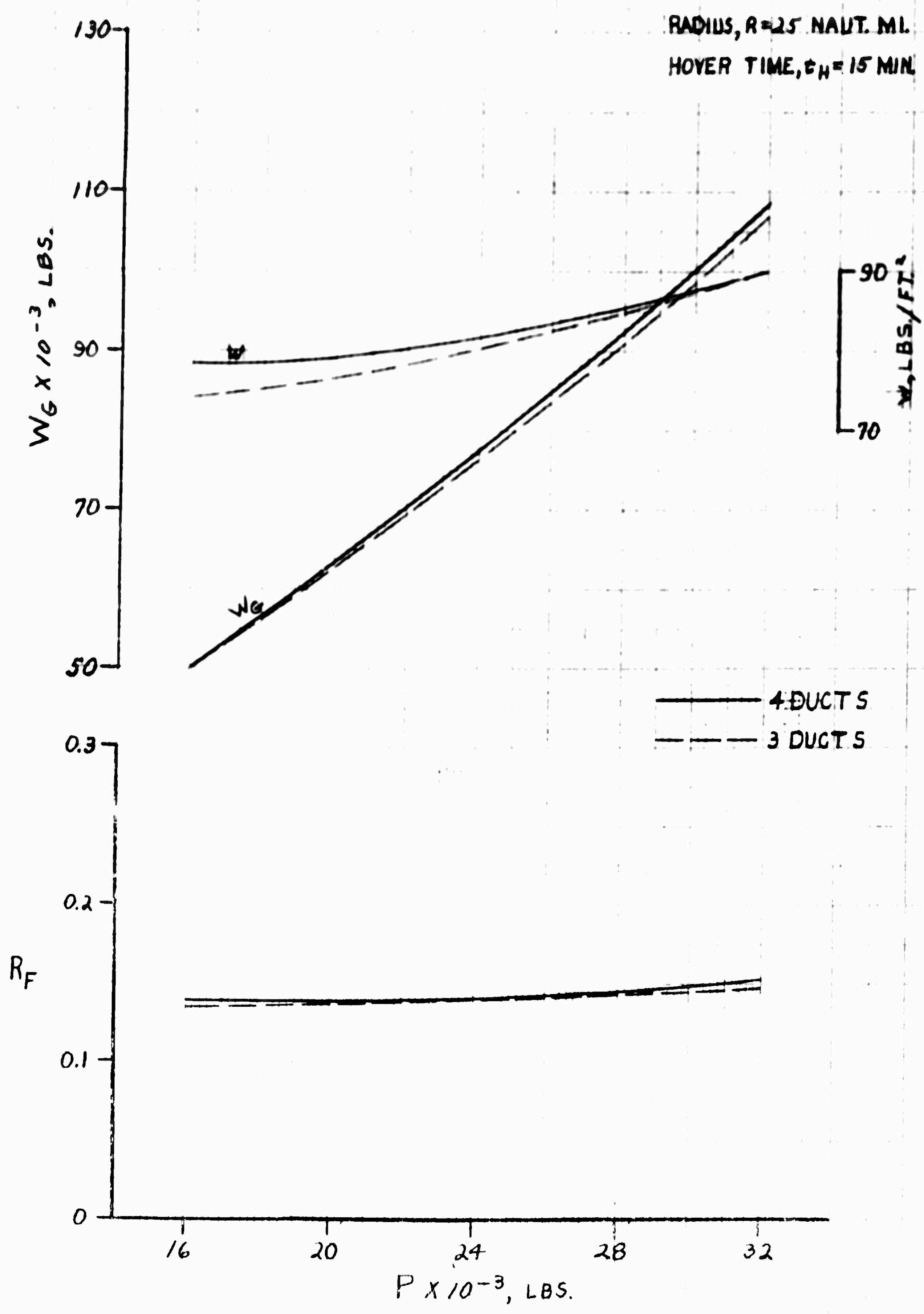
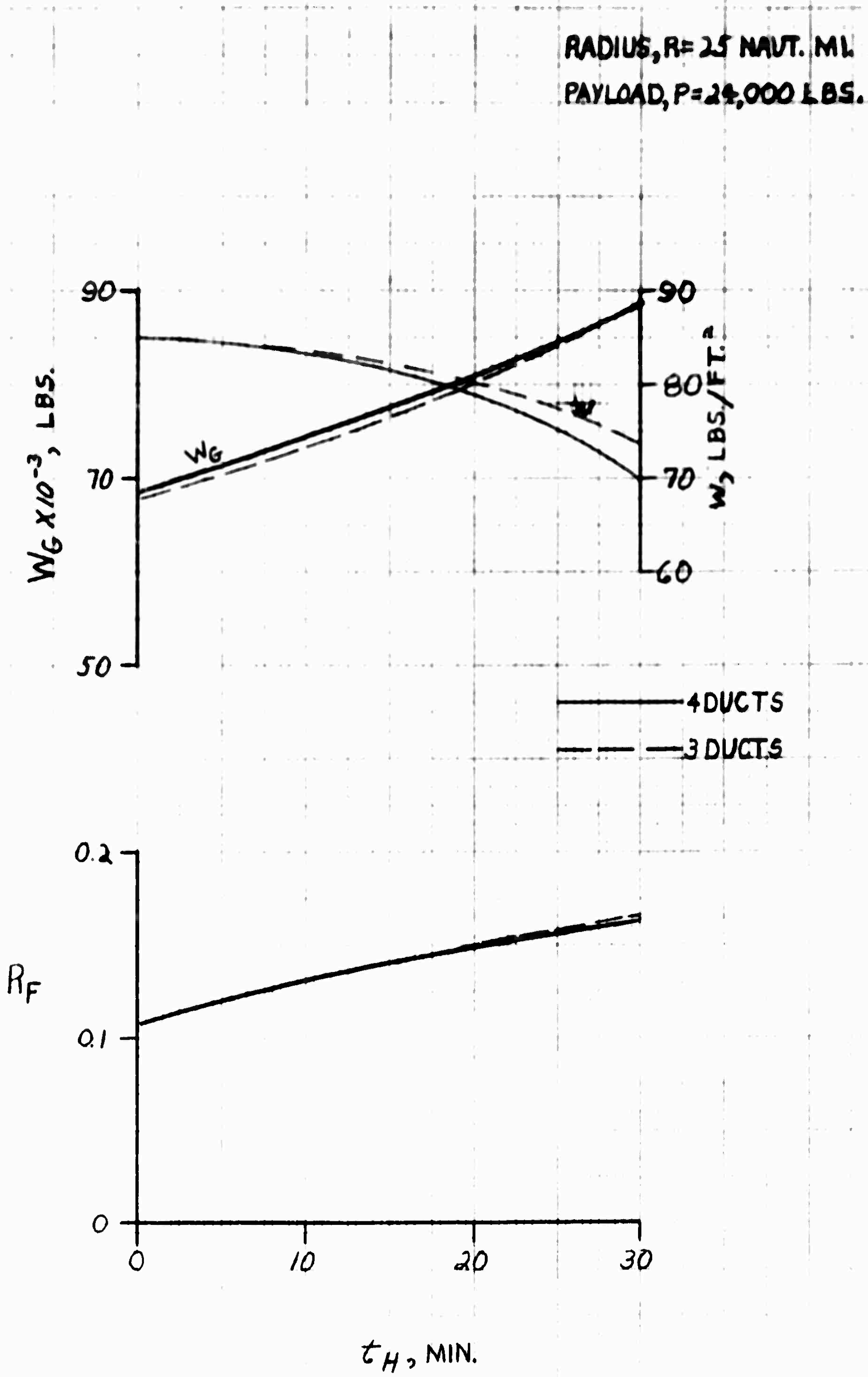
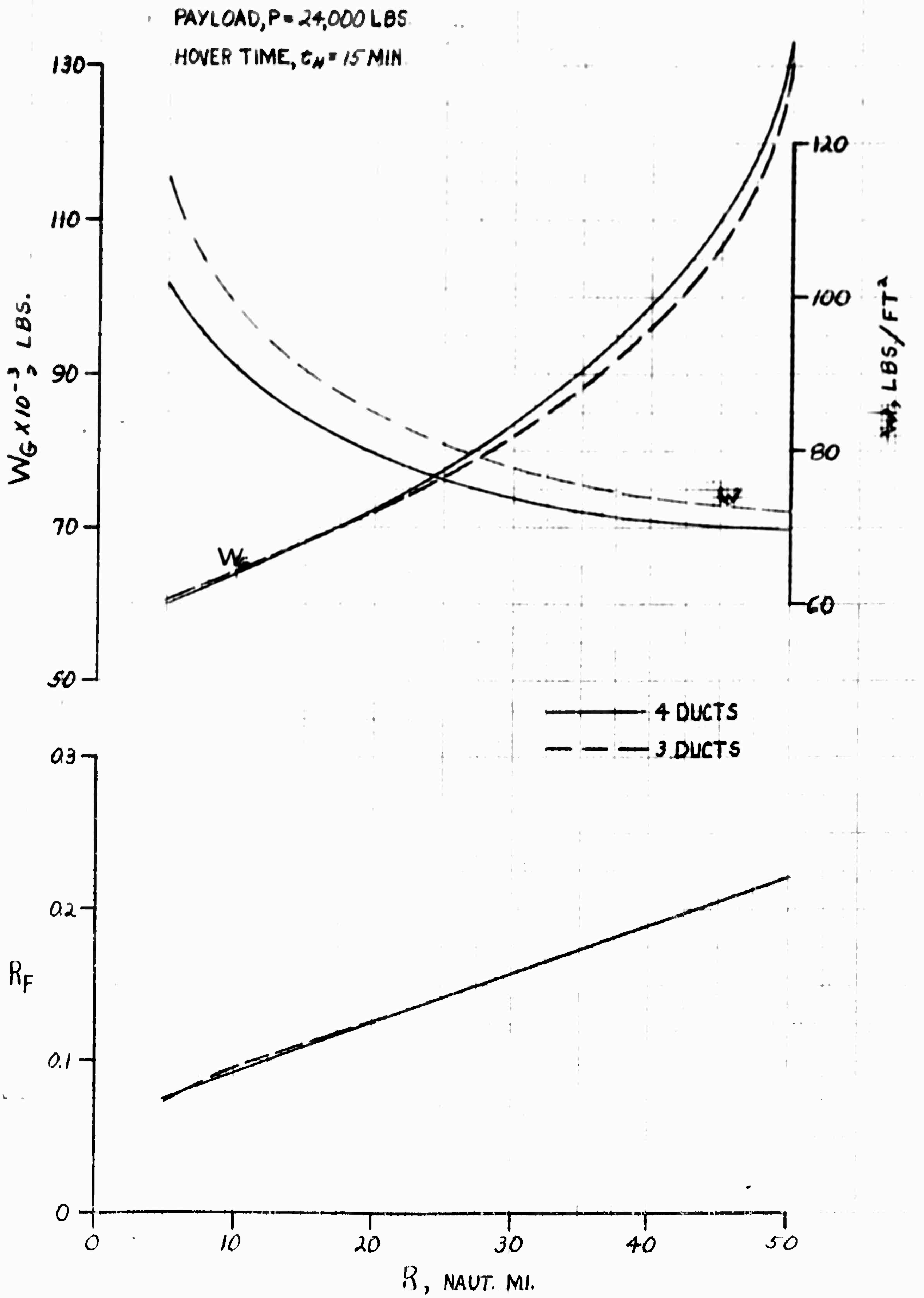


FIGURE 31

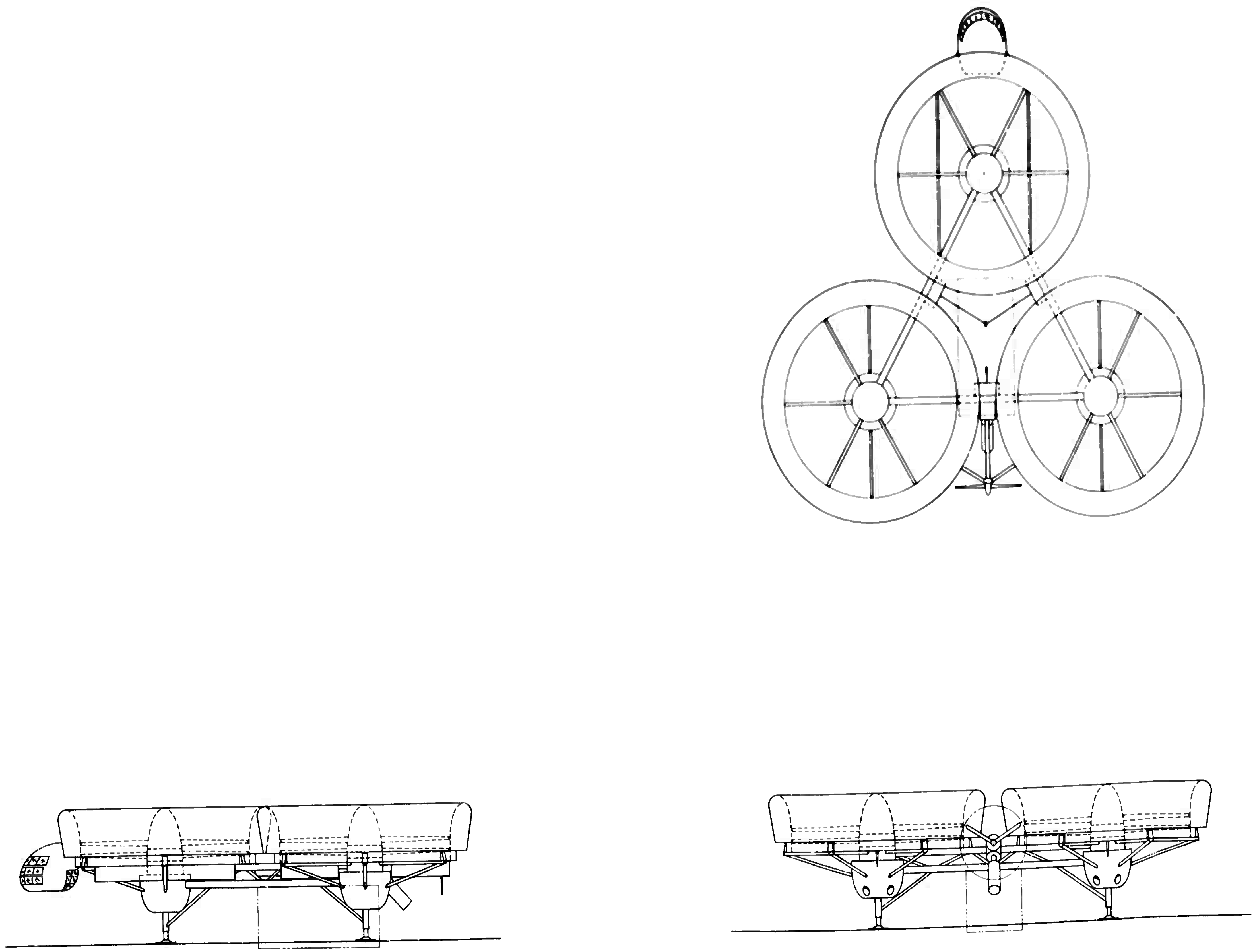
OPTIMUM SHIPS
EFFECT OF HOVER TIME



OPTIMUM SHIPS
EFFECT OF RADIUS

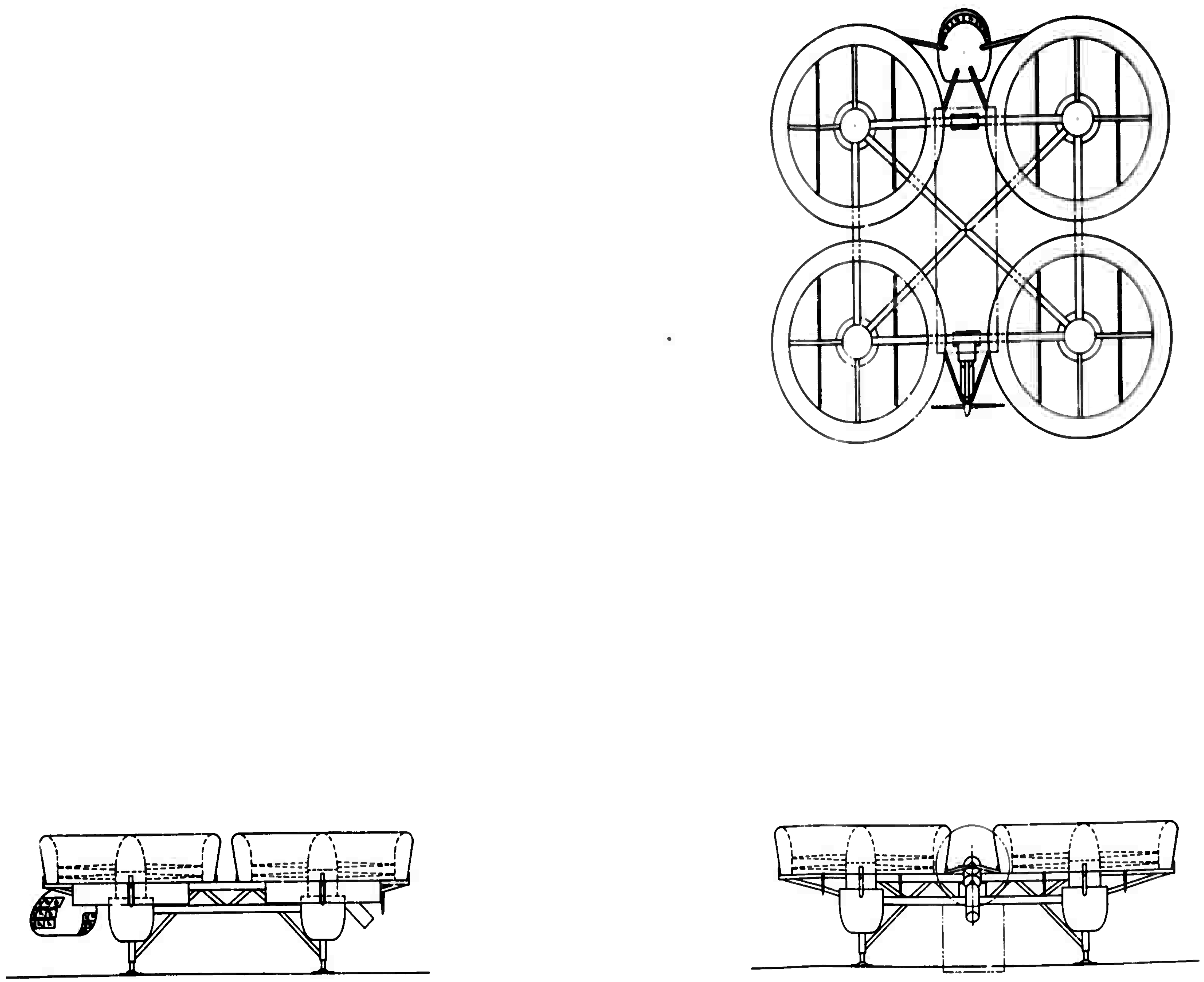


CONFIDENTIAL



Hiller Helicopters
PALO ALTO, CALIFORNIA

THREE - DUCT FLYING CRANE			
SCALE NONE	DRAWN D.L. COLEMAN / APPV'D	DATE 9-24-56 / DATE	DRAWING NO. 1



Hiller Helicopters
PALO ALTO, CALIFORNIA

FOUR-DUCT FLYING CRANE			
SCALE NONE	DRAWN D.L. COLEMAN / APP'VD	DATE 9-24-56 / DATE	DRAWING NO. 2